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FINAL ENGINEERING REPORT

ESSA 1 (TIROS OT-3)

METEOROLOGICAL SATELLITE SYSTEM

VOLUME I

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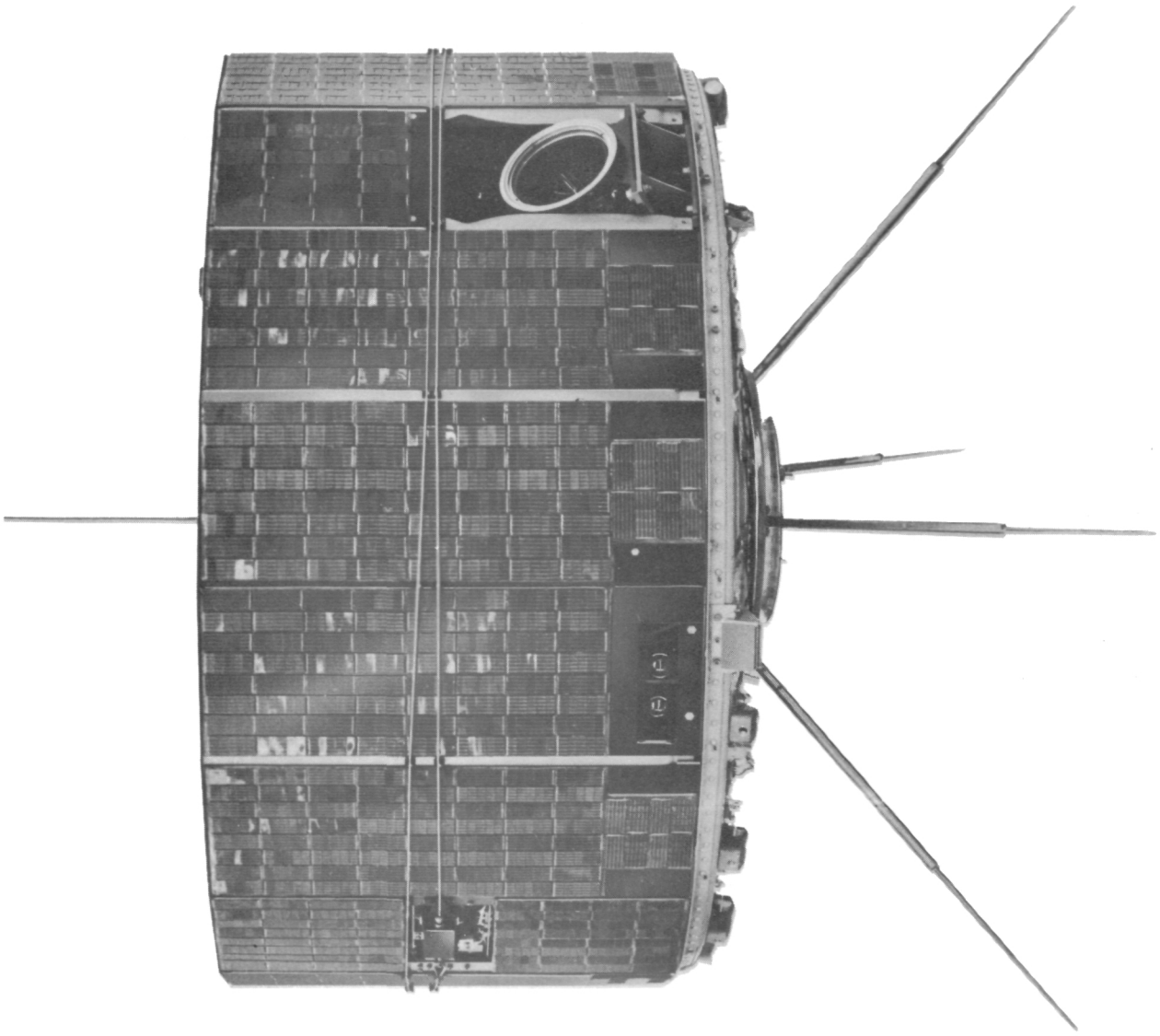

The ESSA-1 Meteorological Sattellite

PREFACE

ESSA 1 was developed and built by the Astro-Electronics Division (AED) of the Radio Corporation of America for the Environmental Science Services Administration (ESSA) under the technical direction of the Goddard Space Flight Center (GSFC) of the National Aeronautics and Space Administration, under NASA Contract NAS5-3173.

This report, which comprises two volumes, is the Final Engineering Report for the ESSA 1 program. The functions of this report are as follows:

(1) To describe the efforts involved in implementing the spacecraft and system design utilized for the ESSA 1 mission.

(2) To provide technical descriptions of the equipment comprising the basic TIROS spacecraft OT-3 configuration and the basic TIROS Command and Data Acquisition (CDA) ground stations.

(3) To describe the efforts involved in determining the ESSA 1 orbital parameters, i.e., those of the near-polar, sun-synchronous orbit such as was first utilized for TIROX IX.

(4) To describe the various system and subsystem tests performed during the program.

(5) To describe the various environmental tests and calibration tests performed on the integrated spacecraft OT-3.

(6) To describe prelaunch and launch activities at the launch site, at the TIROS Technical Control Center (TTCC), and at the CDA stations.

As noted previously, the ESSA 1 final report is divided into two volumes. Volume I is comprised of Parts I through IV. PART I serves as an introduction to the ESSA 1 program and describes the systems concept upon which the program was based, the launch and orbit considerations, and the photocoverage available with ESSA 1. PART II presents technical descriptions of the satellite subsystems and details the various phases of testing performed both on individual units and systems and on the integrated spacecraft. PART III covers the CDA ground station equipment, while PART IV covers the prelaunch and launch operations at the Eastern Test Range, the TIROS Technical Control Center, and the CDA ground stations.

Volume II comprises PART V of the ESSA 1 final report and contains a description of both the satellite-borne and ground-station portions of the programming and control subsystem.

In general, the ESSA 1 final report presents only brief functional descriptions of those spacecraft subsystems and subsystem elements which are essentially the same as their counterparts on previous TIROS satellites. Greater detail on individual elements can be obtained from the respective final reports on the TIROS programs for which the elements were first designed and utilized.

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PART I. INTRODUCTION

SECTION I. PROGRAM SUMMARY

On February 3, 1966, ESSA 1, the TIROS spacecraft OT-3, was launched and orbited from the Eastern Test Range. Successful operation of all subsystems on the satellite was verified on the early orbits of ESSA 1, marking the eleventh consecutive success in the eleven TIROS launches. ESSA 1, the second wheel-type meteorological satellite, provides contiguous coverage of the earth on a daily basis, and will be used in conjunction with ESSA 2 (the TOS APT satellite) in providing the United States with the world's first operational global satellite system.

Spacecraft OT-3 was originally configured as a standard TIROS Series III spacecraft. (Spacecraft OT-1 (TIROS X), OT-2, and OT-3 were procured by the Department of Commerce as Operational TIROS (OT) spacecraft under the technical direction of GSFC.) In 1963, under NASA contract NAS5-3344, AED conducted detailed design analyses covering the conversion of a standard TIROS spacecraft into a wheel-mode spacecraft equipped with either (1) two standard TIROS cameras or (2) one standard TIROS camera and one automatic picture transmission (APT) camera. Based on the results of these analyses, NASA amended the TIROS Series III contract, NAS5-3173, to provide for the reconfiguration of both spacecraft "I" and spacecraft OT-3 into wheel-mode spacecraft utilizing two standard 1/2-inch vidicon TIROS television cameras. This amendment also called for the development of compatible CDA-station and TTCC equipment.

Wheel satellites differ from standard TIROS satellites basically in respect to their mission attitude and the manner in which the cameras view the earth. Both wheel satellites and standard TIROS satellites are spin-stabilized and are injected into orbit with the spin axis in the orbital plane. The spin axis of a wheel satellite, however, is maneuvered until it is normal to the orbital plane; and the cameras are radially mounted to view the earth through openings in the sides of the structure. The spin axis of a standard TIROS satellite remains in the orbital plane, and the cameras are axially mounted and view the earth through the baseplate of the satellite.

Spacecraft "I" was successfully orbited on January 25, 1965, as TIROS IX, the first wheel satellite. Spacecraft OT-3 was assembled, debugged, and qualified as a back-up to spacecraft "I". With the successful orbital operation of TIROS IX, NASA directed that AED calibrate and partially qualify spacecraft OT-3, then place the spacecraft in safe storage in a "hold" condition. On February 12, 1965, the "hold" on spacecraft OT-3 was initiated. Throughout the duration of the hold period, spacecraft OT-3 was subjected to bi-weekly check-out tests. On November 18, 1965, by

NASA directive, spacecraft OT-3 was removed from the "hold" condition for the remaining qualification program and final calibration and prepared, with a target date of January 1966, for delivery to the Eastern Test Range.

The AED activity during the ESSA 1 program included the evaluation of the orbital performance of TIROS IX and TIROS X to determine if any modifications to spacecraft OT-3 were required for the ESSA 1 program. The modifications made as a result of this evaluation and as a result of the ESSA 1 test program included the following:

- A sun shield was added to the view ports for the horizon scanners to reduce the possibility of sun interference.
- RF isolation of the beacon transmitters was improved by the addition of bandpass filters and attenuators in the RF coupling system.
- A new 4-port hybrid was added in the command receiver harness.
- The length of the magnetic tape in the tape recorders was increased to 830 feet and new, 4-foot, silvered leaders were added.
- The telemetry commutators and the Dycon units were slightly modified.

Special tests were performed at AED in the course of the ESSA 1 test program. These tests included the following:

- Mechanical and electrical checks of the combined spacecraft and model third-stage launch vehicle.
- Leak rate tests on the tape transports, and life tests on the wiring insulation in the transports.
- A systems test of beacon-transmitter operation.

The further responsibilities of AED during the ESSA 1 program included the following:

- Qualify the subsystem elements required for spacecraft OT-3, and provide qualified spare units as back-up equipment.
- Install the qualified subsystem elements on an appropriately modified TIROS structure, integrate the subsystem into a complete spacecraft system, and ensure that the integrated spacecraft met the established specifications.
- Perform the required environmental tests on the integrated spacecraft to ensure that it would withstand the rigors of launch and the planned orbital environment.

- Calibrate the spacecraft's sensing elements to ensure the capability of accurate attitude control and photo-coverage predictions.

- Upon satisfactory completion of the environmental testing and final calibration, deliver the spacecraft to the ETR for launching, and provide engineering support for the launch effort.

- Perform prelaunch analyses of the launch and orbit requirements.

- Prepare instructions for establishing and maintaining the wheel attitude.

- Prepare instructions for checkout of the spacecraft's functions and evaluation of the spacecraft's performance during the initial orbits.

- Prepare instructions for commanding the satellite for the performance of specific attitude-control, spin-rate control, and data-gathering functions.

- Prepare instructions for receiving and processing data from the spacecraft, and for maintaining the ground equipment in good operating condition.

- Provide direct engineering support to the CDA ground stations and TTCC during the launch and immediate post-launch periods.

SECTION II. SYSTEM DESCRIPTION

A. THE ESSA 1 SATELLITE

The key sensors on ESSA 1 are two standard TIROS 1/2-inch vidicon TV cameras utilized in the two identical camera chains which make up the TV-picture subsystem. Each camera chain also includes a complete recording system, video processing circuits, a transmitter, and has, as associated equipment, two alternate units for camera triggering, i.e., a computing trigger and a Dycon. The operation of both types of camera-triggering units is dependent on the pitch geometry, as determined by the placement of horizon scanners on the baseplate relative to the cameras. As noted, the cameras view the earth through the sides of the structure, and are mounted at diametric positions on the baseplate. The cameras are canted at equal, but opposite, angles with respect to the baseplate, and, as the satellite progresses in its orbit, camera No. 1 "looks" eastward and camera No. 2, westward.

Dynamics control is provided, in part, by the despin devices, precession dampers, and spin-up rockets. In addition, ESSA1 utilizes the magnetic-torquing techniques first developed for TIROS IX and proven in orbit in the TIROS IX mission. Two torquing coils, the Quarter Orbit Magnetic Attitude Control (QOMAC) coil and the Magnetic Bias Control (MBC) coil*, are used for orientation and control of the spin-axis attitude; while a third, the Magnetic Spin Control (MASC) coil, is used for control of the spin rate.

In addition to the two pairs of horizon scanners, which supply an indication of absolute pitch for camera triggering, the attitude-measurement devices on ESSA 1 include the V-head attitude sensor and the digital solar-aspect indicator. The V-head attitude sensor provides continuous data from which the attitude of the satellite can be determined; while the digital solar-aspect indicator provides a measurement of the gamma angle, i.e., the angle between the satellite-sun line and the positive spin axis of the satellite. The primary means of attitude determination is from an analysis of the data supplied by both channels of the V-head attitude sensor.

The data obtained by the attitude-measurement devices, plus "housekeeping" data covering various satellite operating parameters, is transmitted to the ground stations

*On the TIROS satellites preceding TIROS IX, beginning with TIROS II, the MBC coil was used alone as the sole means of attitude control.

by means of two beacon transmitters in the telemetry and tracking subsystem. The beacon transmitters supply continuous signals to facilitate tracking and provide four subcarrier channels for telemetry data. Two subcarrier channels are used continuously for the transmission of V-head attitude sensor data, and two are used, on command, for back-up attitude data, "housekeeping" data, horizon-scanner data, and solar-aspect-indicator data. The "housekeeping" data is supplied by two commutators which provide redundant sampling of 90 data points.

The satellite's picture-taking and dynamics-control operations at remote locations are controlled by means of a programming and control subsystem, which receives, decodes, and stores ground-initiated commands and activates the components required to perform the commanded function.

Both cameras on spacecraft OT-3 are required in order to complete the mission. However, some elements in the two identical supporting chains are redundant, and the operation of a camera does not depend on the full operation of each unit in the associated supporting chain. The TV transmitters, the camera-triggering units, and the horizon scanners are redundant. In addition, the camera clocks (in the programming and control subsystem) can be cross-coupled, by command, to permit either or both camera systems to be operated by either clock. The tape recorders can also be cross-coupled in the playback mode to permit sequential operation of both recorders by a single command.

Selective redundancy is available with Dycon operation in the dynamics-control subsystem, while redundant operation is automatically provided by the telemetry and tracking subsystem. In the latter, the same information is transmitted over both beacon subcarriers though in different sequences.

Power for the satellite's operating requirements is supplied by means of a solar-energy converter operating in conjunction with a group of nickel-cadmium storage batteries. During orbital day, the electrical requirements of the satellite are supplied by the solar-energy converter, and any excess power available is used to charge the storage batteries. During orbital night, all power requirements are supplied from the battery pack.

Structurally, ESSA 1 is almost identical to the ten TIROS satellites which preceded it. The structure comprises a baseplate, 42 inches in diameter, upon which the subsystem components are mounted, and a "hat", 18 inches in height, which covers the baseplate and serves as a mounting area for the solar cells.

B ORBITAL PARAMETERS

Inasmuch as a daily global photocoverage was the basic consideration of the spacecraft OT-3 mission, the orbital parameters were chosen to optimize the scene-illumination

characteristics, the area coverage per picture, the contiguity of successive pictures, and the picture resolution. The design goals for this satellite were identical to those for TIROS IX, and were as follows:

- altitude: 400 nautical miles,
- shape: circular,
- inclination: 98.36 degrees (81.64 degrees, retrograde),
- nodal period: 99.628 minutes,
- earth rotation per nodal period: 24.6 degrees, and
- percentage of sun time: 70 percent (average).

These parameters define a near-polar, sun-synchronous orbit which provides optimum utilization of the wheel satellite and permits total earth coverage on a regular, recurring basis and in a minimum time. It should be noted that the considerations affecting the choice of an orbital altitude also included (1) the use of the Delta DSV-3C launch vehicle and (2) the fact that a spacecraft launched southward into a retrograde orbit from the Eastern Test Range (ETR) must be guided in a "dogleg" trajectory around the tip of Florida.

The mechanics of the sun-synchronous orbit are the same as those of any circular earth orbit, with the unique exception that the orbital plane of the sun-synchronous orbit revolves around the earth's polar axis in the same direction and at the same rate as the earth-sun line. This nodal precession results in a nearly constant angle between the orbital plane and the direct rays of the sun, so that the satellite's subtrack on the surface of the earth receives nearly the same solar illumination on every orbit throughout the year.

An ascending node daylight crossing was chosen for the ESSA 1 mission to optimize the speed with which remotely taken pictures could reach a ground station for processing and use. Though the same total coverage is obtained regardless of the direction in which the satellite crosses the nodal point during the daylight parts of the orbit, the determining factor in choosing an ascending daylight crossing was the fact that the Wallops Island, Virginia, and Fairbanks, Alaska, CDA ground stations could read out the pictures recorded during the daylight ascending pass.

The launch time was scheduled so as to establish a time of between 1400 and 1500 hours for the daylight crossing and, thereby, provide satisfactory mission-mode gamma angles, since the time of daylight crossing, for a given time of year, fixes

the gamma angle. This choice, in turn, ensured satisfactory power output and temperature conditions for the ESSA 1 mission.

Further details on the various factors affecting the choice of launch time and direction of the daylight crossing can be found in the TIROS IX Final Engineering Report.

C. GROUND COMPLEX

The ESSA 1 ground complex includes two primary (CDA) stations: one located at Wallops Island, Virginia, and the other at Fairbanks, Alaska. These stations are the same facilities used on previous TIROS programs, and, as such, are compatible with the requirements of both the wheel satellites, such as ESSA 1 and TIROS IX, and the standard TIROS satellites.

ESSA 1 operations are controlled from the TIROS Technical Control Center (TTCC) at NASA's Goddard Space Flight Center in Greenbelt, Maryland. This facility is operated in conjunction with the following facilities: The NASA Space Operations Control Center, the NASA Space Computer Center, the Minitrack Network, the TIROS Film Control Office, and the National Environmental Satellite Center of the Environmental Science Services Administration.

The satellite programming for each orbit of ESSA 1 is formulated by TTCC, based upon requests for information from the other agencies participating in the program. The program for each orbit is teletyped to the CDA stations along with detailed ephemeris data, which is prepared by the Space Computing Center. The ephemeris data is used for orienting the CDA station antennas to ensure early satellite acquisition, resulting in maximum contact-time.

The CDA station transmits the command program to the satellite in the form of radio-frequency signals. In turn, the station (1) receives the radio signals carrying the television, attitude, and telemetry data from the satellite; (2) processes, records, and labels the received data; and (3) transmits it to TTCC and to the facilities associated with the ESSA 1 ground complex.

The TV photographs returned by the satellite are recorded at the ground station on magnetic tape and on photographic film; other data from the satellite is recorded on paper charts. The film, tape, and paper-chart records are processed and identified according to date, satellite name, and orbit number before being shipped to TTCC (through the Space Operations Control Center). Negatives of the cloud-cover photographs taken by the satellite are also delivered to the TIROS Film Control Office and to the National Environmental Satellite Center. In addition, copies of the photographs are retained at the CDA stations where they are used, by meteorologists assigned to the stations, in preparing nephanalyses for transmission, via photofacsimile, to authorized agencies in various parts of the world.

In addition to the primary CDA stations, a back-up and evaluation station is located at the Astro-Electronics Division of RCA, near Princeton, New Jersey. Stations of the NASA STADAN Network are also included in the ESSA 1 ground complex; however, these stations do not actively participate in satellite programming, but serve only to provide accurate data on the satellite's orbital parameters. The data gathered by the STADAN Network and related data from the NASA Computer Center are primary factors in determining the programming for each satellite orbit.

SECTION III. SYSTEM OPERATION

A. DYNAMICS CONTROL

Components of the dynamics-control subsystem are provided (1) to control the spacecraft when it is first injected into orbit, (2) to orient the spacecraft to the wheel attitude, and (3) to maintain the proper attitude and spin rate during the operational life of the satellite.

When the satellite is separated from the third-stage of the launch rocket, i.e., approximately 1.5 minutes after the combined assembly has been injected into orbit, the lift-off and separation switches close (an indication of separation is telemetered to the ground), the precession dampers are automatically activated, and the timing for the automatic activation of the despin mechanism is initiated. The precession dampers are tuned-energy-absorption-masses which cancel out any forces occurring at separation which might tend to nutate the satellite.

When the satellite third-stage rocket assembly earlier separated from the second-stage rocket, the combined assembly was spun up at approximately 125 rpm. Approximately 9.5 minutes after the satellite has separated from the third-stage rocket, the despin weights are automatically released. The despin cables, with the attached weights are wrapped around the satellite's solar hat, approximately at midpoint. When the weights are released, the cables unwind, and the cables and attached weights are cast off from the satellite after absorbing sufficient energy to cause despin. The despin takes place within approximately 0.5 second, reducing the spin rate to the 8 to 12 rpm range.

Though both the precession dampers and the despin mechanism are designed for automatic operation, a back-up capability for ground-controlled activation is also available.

When the satellite is first injected into orbit the spin axis lies nearly in the orbital plane. Before picture-taking operations can be initiated, the satellite must be reoriented so that the spin axis is normal to the orbital plane. This is accomplished within the first day of operation, so that the desired sun angle will be achieved while the operating temperatures of the satellite components are within the design limits, and adequate power is available from the storage batteries.

The wheel maneuver is accomplished by means of the torque produced on the satellite as a result of the interaction of the earth's magnetic field and the field created around

the satellite when currents are caused to flow in the QOMAC coil. The QOMAC coil is capable of operation in both the high- and low-torque modes, and the high-torque mode is used in the initial wheel maneuver.

A computer simulation of the wheel maneuver was performed before launch, and the results of this simulation are summarized in Figure I-1.

After the first indications of satellite attitude have been received at the CDA stations, the required number of QOMAC cycles for accomplishing the wheel maneuver is confirmed and the torquing program is transmitted to the satellite.

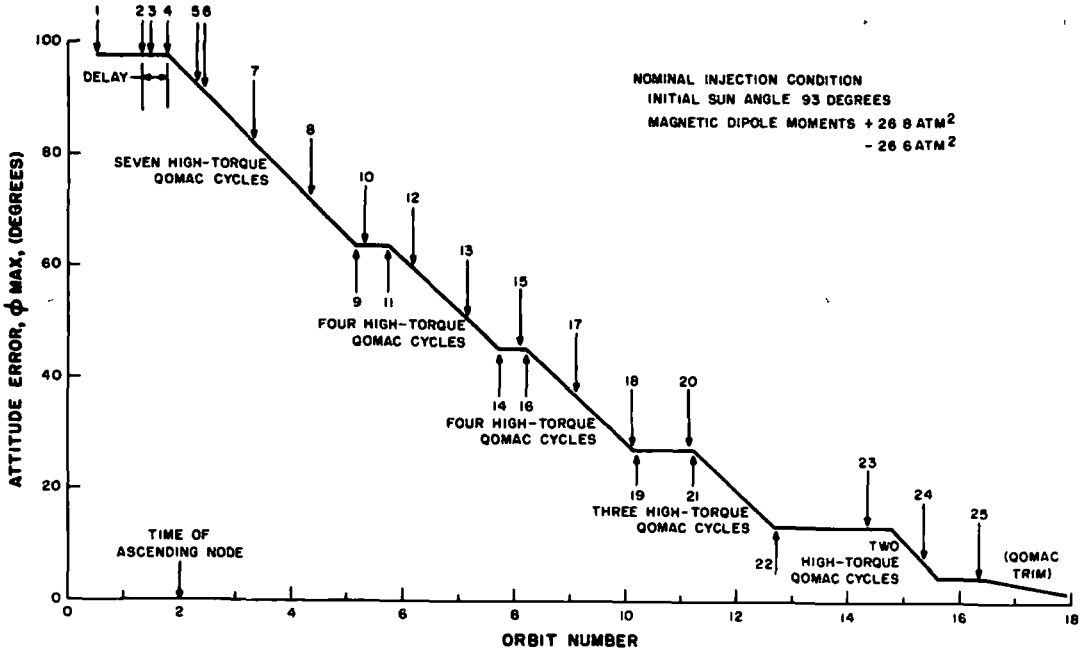
The first QOMAC torquing cycle is initiated on orbit 0001, and on each pass during the course of the maneuver, data is continually obtained and evaluated in order to monitor the progress of the maneuver. During this time, the exact orbit parameters are determined, along with the status of the telemetry and tracking subsystem, the programming and control subsystem, and various other components within the spacecraft.

After the wheel attitude has been achieved, the spin axis is permitted to drift for several days so that the satellite's residual magnetic dipole moment can be measured Based on this measurement, a constant torque is programmed in the MBC coil to negate the residual dipole moment. An additional torque is then programmed in the MBC coil to compensate for the 1-degree per day precession rate of the orbital plane in a sun-synchronous orbit.

As the satellite continues in its operational life, the low-torque QOMAC cycles can be used to maintain the spin axis in the wheel attitude In addition, fine adjustments of the spin rate are available by means of the MASC coil, whereas spin-up rockets are provided for redundancy.

Use of the MASC coil permits close maintenance of the spin rate at a specific value for long periods of time. In a MASC cycle, current is programmed through alternate halves of the coil every half spin of the satellite. The resultant torque may be applied at quarter orbit intervals, so that the current flows through the coil only while the satellite is over the earth's magnetic poles, where the field intensity of the earth is greatest. By controlling the direction of current flow in the coil, the spin rate can be either increased or decreased.

Because of this capability for controlling the spin rate by means of the MASC coil, it was possible to use the Dycons in spin-synchronized camera triggering, thereby offering a back-up means of shutter control in addition to the use of the computing triggers.



LEGEND

1 Satellite launch

2 First QOMAC program transmitted to satellite from Wallops Island CDA Station during orbit No. 0001

3. Telemetry readout (Fairbanks CDA Station)

4 DYCON No. 2 alarms, first high-torque QOMAC program commences at 75 minutes after ascending node (AAN), orbit No. 0001

5 Telemetry readout

6 Telemetry readout

7 Telemetry readout

8 Telemetry readout

9 First QOMAC program ends at 25 minutes AAN, orbit 0005

10. Second QOMAC program transmitted to satellite from Fairbanks CDA Station during orbit No 0005 Telemetry readout

11 DYCON alarms, second high-torque QOMAC program commences at 75 minutes AAN, orbit No 0005

12 Telemetry Readout

13 Telemetry Readout

14 Second QOMAC program ends at 75 minutes AAN, orbit No 0007

15 Third QOMAC program transmitted to satellite during orbit No 0008 from Fairbanks CDA station Telemetry readout

16 Third high-torque QOMAC program commences at 25 minutes AAN, orbit No 0008

17 Telemetry readout

18 Telemetry readout

19 Third QOMAC program ends at 25 minutes AAN, orbit No 0010

20 Fourth QOMAC program transmitted to satellite from Fairbanks CDA station during orbit No 0011

21 DYCON alarms, fourth high-torque QOMAC program commences

22 Fourth QOMAC program ends at 75 minutes AAN, orbit No 12

23 Fifth QOMAC program transmitted to satellite from Wallops Island CDA station during orbit No 0014 Telemetry readout

24 Telemetry readout

25 First low-torque QOMAC program transmitted to satellite from Fairbanks CDA station during orbit 0017

Figure I-1. Simulated Initial Orientation Maneuver for ESSA 1

B. CAMERA OPERATION

1. General

Four modes of operation are available with the TV-picture subsystem: direct, remote, direct-remote, and playback.

Direct-picture requests are transmitted to the satellite while it is in contact with the CDA station. The transmitted command specifies the camera system to be used and the camera-triggering unit, which, using horizon information from the horizon scanners associated with the selected camera, controls camera operation so that the shutter is actuated when the camera view axis is aligned with the local vertical. The resulting video signals are read out in real time, processed, and applied by means of the video coupler to one of the two, frequency-modulated TV transmitters for direct transmission to the ground.

The majority of pictures to be taken during the ESSA 1 mission, however, will be remote pictures, since the satellite is within communications range of the CDA stations for only a relatively short portion of each orbit. While the satellite is in contact with a ground station, remote-picture sequences are programmed into one of the two digital clocks in the satellite, permitting TV coverage of an area of interest remote from the ground stations.

The commands programmed into the digital clock consists of a five-part message that conveys the required values of five variables in the remote sequence. These variables are as follows:

(1) The remote-sequence delay time: the interval which must elapse between the command and the time that the first sequence is started. This interval can be programmed, in 2-second increments, over a range from 2 seconds to 18 hours, 7 minutes, and 12 seconds.

(2) The remote-sequence duration: can be programmed for the following durations: 17 minutes, 4 seconds; 25 minutes, 36 seconds; 34 minutes, 8 seconds, and 42 minutes, 40 seconds.

(3) The picture interval: the time that elapses between picture requests in the remote sequence. Intervals of 32, 64, and 128 seconds can be programmed. The picture interval, in conjunction with the remote-sequence duration, determines the number of pictures that will be taken in a remote sequence. (The actual time duration between pictures will be modified slightly by the computations of the camera-triggering units.)

(4) The number of sequences to be taken: the picture-taking sequence can be programmed to be repeated at the same point in consecutive orbits without an intervening satellite-to-ground contact. This feature is made possible by an orbit-timing capability built into the camera clocks. A total of four consecutive sequences can be programmed.

(5) Orbital-period correction factor: used to compensate for the deviation of the actual orbital period from the predicted period (102 minutes, 24 seconds) wired into the clocks before the satellite is launched. The correction can reduce the pre-wired period by as much as 4 minutes, in 16-second increments. (Once the orbital parameters have been determined accurately, the correction factor will remain constant. It must, however, be transmitted each time the clock is programmed.)

The direct-remote mode is a hybrid of the direct and remote modes. Because of the relatively long remote-picture-taking sequences available, it is possible that the satellite will be in communications range of a CDA station while it is taking remote pictures previously programmed by another station. Although the ground station cannot originate direct-picture requests during this type of contact without interrupting the remote sequence already programmed into the satellite, it can receive the pre-programmed pictures as they are taken and recorded on the camera system's tape recorder. This is accomplished by transmitting a continuous direct-picture tone to the satellite. Since the camera and associated electronics are already powered by the digital clock, the effect of the direct tone is to turn on one of the satellite's TV transmitters. Then, as the remote pictures are recorded in the satellite, they are also transmitted to the ground station.

The playback mode is commanded, while the satellite is in contact with the CDA station, to obtain the information obtained in a remote sequence and stored on the satellite's tape recorders. Playback is effected by the transmission to the satellite of a Playback 1 or 2 tone (depending on the camera system being interrogated). After the satellite has received the tone for the proper interval, power is applied to the selected tape recorder and recorder electronics and, through the digital auxiliary control unit, to the selected TV transmitter.

A cross-coupling feature is available with the tape recorders, so that when the proper digital auxiliary command is transmitted both recorders will play back sequentially in response to one command.

2. Direct Mode

After the satellite has been properly addressed, and either a Direct 1 or Direct 2 command tone has been transmitted, the command-control unit in the programming and

control subsystem delivers warm-up power to the commanded camera system and turns on the appropriate TV transmitter. After the warm-up is complete, the command-control unit delivers power to the remainder of the camera system and to the auxiliary control unit.

For example, if a Direct 1 tone is transmitted (and the digital clocks are not cross-coupled), an enabling voltage from camera-control unit No. 1 is applied to the camera-power switch in camera clock No. 1. This switch energizes and delivers warm-up power to the camera No. 1 electronics and camera-triggering units. Power for the TV transmitter is applied through a relay in the digital auxiliary control unit to either (but not to both) of the TV transmitters. The position of this relay controls which of the transmitters will be energized and depends upon the last transmitter-select command transmitted to the satellite.

A momentary interruption in the direct-camera tone acts as a command for a direct picture. Each time an interruption occurs, a picture-request pulse is delivered to all four camera-triggering units associated with the TV-picture subsystem. If the Dycon unit associated with the commanded camera system has been placed into the camera-triggering mode by ground command, an inhibiting level will be applied from that unit to the computing trigger, and only the Dycon will act on the picture request. If the Dycon is not in the camera-triggering mode, the computing trigger will perform the camera-triggering operation. (Since the Dycon and computing trigger on side 2 are not energized by a Direct 1 tone unless the camera clocks are cross-coupled, neither unit on that side will act on the picture request.)

After receiving the request, the activated camera-triggering unit, using spin rate and pitch information from the associated horizon scanners, begins to generate vertical sync pulses which are applied to camera No. 1. These pulses, which are phased to coincide with shutter actuation, are used, in conjunction with horizontal sync pulses generated by the camera electronics unit, to generate sweep voltages prior to shutter actuation, thereby erasing the vidicon and preparing it for exposure. The camera-triggering unit also anticipates the time at which camera No. 1 will be looking downward at the satellite subpoint, and, at that time, generates a shutter pulse to expose the vidicon and to turn on the video subcarrier oscillator (VSO) in the recorder electronics unit.

During the 2-second interval following vidicon exposure, the image on the vidicon is read out by the camera electronics, and the resulting video signal is applied as frequency-modulation to the VSO. The resulting video subcarrier is applied through the tape-recorder electronics, bypassing the tape recorder, to the video coupler, and is used to frequency-modulate the activated TV transmitter. The resulting FM/FM modulated RF carrier is transmitted directly to the interrogating CDA ground station.

After the readout of the vidicon has been completed, the activated camera-triggering unit prepares for the next picture request. Should a second picture request occur before the first command has been acted upon, the second request will be ignored.

3. Remote Mode

The satellite is programmed for remote operation during a satellite-to-ground contact only during the period when a Playback 1 or Playback 2 mode is being executed. The ground station performs the remote-picture programming operation by interrupting the playback tone for several seconds and by transmitting an FSK digital command word to the satellite. The command word for programming remote-camera operation contains 34 binary data bits. Of these, the first four are used by the programming and control subsystem to enable the command decoder; and the next five, a unique channel-select code, cause the decoder to enable either digital clock No. 1 or digital clock No. 2.

In the case that the decoder has enabled digital clock No. 1, the remaining 25 bits of the 34-bit command word are then shifted into digital clock No. 1, and contain the actual instructions for the remote picture-taking sequence.

The digital clock is basically a 25-stage, serial, flip-flop register equipped with coincidence gates and power-switching circuits. During the period that the 25 bits of the command word are being loaded into the clock, the entire register functions as a serial-shift register. After loading, part of the clock functions as a counter while other sections store parts of the command read into it. The digital clock commences counting out the programmed delay time as soon as the loading operation is completed, using 0.5-pps pulses from the time-base generator (in the programming and control subsystem) as clock pulses.

When the programmed delay time has been counted out, power is delivered to the camera, to the camera electronics, to the selected camera-triggering units, and to the tape-recorder electronics. Picture requests are simultaneously delivered to all four camera-triggering units at the programmed interval, and the activated camera-triggering unit begins computing the shutter-actuation time. After receiving each request, the activated camera-triggering unit (1) delivers pulses to prepare the vidicon for exposure, (2) starts the tape-recorder transport, and (3) actuates the camera shutter and turns on the video subcarrier oscillator in the recorder electronics unit.

After exposure, the image on the vidicon is read out and applied as modulation to the VSO. The output of the VSO is recorded on channel 1 of the tape transport. During the 2-second readout period, a Picture Time Delay Code (PTDC) is generated in the activated camera-triggering unit and applied as pulse modulation to a 10-kc subcarrier oscillator located in the recorder electronics unit. This signal, which is

indicative of the time elapsed between a picture request and the shutter command, causes pulse-code modulated bursts of the 10-kc frequency to be recorded on channel 2 of the tape transport during the same period that the video information is recorded on channel 1.

At the end of the 2-second readout period, the output from the activated camera-triggering unit which is used to power the tape-recorder transport is terminated, and the camera-triggering unit recycles continuously until the next picture request is received, at which time the transport is again powered. Picture requests continue to be initiated by the digital clock, at the specified interval, until the completion of the last picture of the programmed remote-picture sequence, at which time power is removed from the camera, the camera-triggering unit, and the tape recorder electronics.

At the start of the remote-picture sequence, the first 15 stages of the register in the camera clock (the stages which contained the programmed delay-time setting) reset, then begin to count the sequence duration. If more than one sequence is programmed, the register continues to count after the sequence ends, counting the time from the start of the first sequence to the start of the second, a time period equivalent to one orbital period. These stages are so wired that when the total count of the 0.5-pps signal represents a time period equivalent to the pre-programmed orbital period, the register will overflow and cause another remote-picture-taking sequence to begin. (The orbital correction factor is added to the count approximately midway in the counting cycle, thereby deducting the time increment necessary to cause this overflow to occur after one actual orbital period.) The remote picture-taking sequence will then repeat, and the camera system will take pictures at the same interval and for the same duration as in the previous sequence, then continue timing for the total number of consecutive sequences programmed, with each sequence starting at approximately the same position in the orbit. At the end of the final sequence, the camera clock is reset, power is removed from it, and the tape transport is enabled for playback.

The capacity of each of the satellite tape recorders is greater than 48 TV pictures. The total number of pictures requested from one camera in a remote program, considering the three variables of picture interval, sequence duration, and number of orbit repeats, cannot exceed this amount.

4. Direct-Remote Mode

During a remote sequence, the programmed camera, camera electronics, camera-triggering unit, and tape recorder are energized by power applied from the camera clock. When the direct tone is received by the satellite to effect the direct-remote mode of camera operation, these units are not affected, since they are already energized; however, the TV transmitter is turned on by the command-control unit. As

each remote picture is initiated by the programmed request, the resultant video subcarrier is simultaneously applied to the TV transmitter as it is recorded by the tape recorder, and, therefore, is transmitted to the interrogating ground station.

Certain precautions are observed at the ground station when interrogating the satellite in the direct-remote mode. Because the picture requests from the camera-control unit and the camera clock are applied at a common input to the activated camera-triggering unit, any interruption in the direct tone from the ground station would be interpreted as a normal picture request, and the picture resulting from this spurious request would be stored on the tape recorder. The additional picture would make interpretation of the pictures on the tape difficult, and could result in overloading of the tape. Since the drop-out of the direct tone at the end of a direct-remote interrogation could also be interpreted as a picture request, the tone is dropped out during the period after a request has been made by the camera clock, but before the end of the 2-second vidicon-readout period, since during this period the picture-requests are ignored by the camera-triggering unit. This can be accomplished by dropping the direct tone at the start of a TV-picture transmission, as indicated by the appearance of the subcarrier on the ground-station monitors. (In those cases where the camera clocks are cross-coupled, the tone must be dropped out at the beginning of the first-received picture of the complementary pair.)

5. Playback Operation

When the camera clock has completed all remote sequences called for in the pre-set program, the transmission of the Playback 1 or 2 tone (depending on the system to be interrogated) will effect playback of the video information stored in the satellite tape recorder. If the camera clock has not completed the programmed remote sequences, the playback tone effecting recorder read-out must be preceded by a digital reset command to the operating camera clock.

After a playback tone has been transmitted for the proper interval, power is applied to the tape recorder and recorder electronics and to the selected TV transmitter. For example, if a Playback 1 tone is the first transmitted to the satellite, playback power is applied from command control No. 1 to tape recorder No. 1 and, through the digital auxiliary control unit, to the selected TV transmitter. When power is applied to the recorder tape transport, the recorder playback is initiated and synchronized by the 500-pps output from the time-base-generator. The 85-kc video subcarrier and the 10-kc subcarrier containing the shutter delay code (stored on channel 1 and channel 2, respectively) are read out and delivered to the recorder electronics unit, where the two subcarriers are mixed. The composite frequency is used to frequency-modulate the TV transmitter.

When playback is completed, the recorder stops automatically, and, since the tape is erased during playback, the unit is ready for the next remote-picture-taking sequence. At this time the ground station normally transmits the Playback 2 tone, and the playback operation is repeated for tape recorder No. 2.

If, during the life of the satellite, a malfunction should develop in the playback-command chain for either tape recorder, the recording system will still be usable because of the cross-coupling feature incorporated into the playback circuits. A camera clock cross-coupling relay controllable by ground-initiated commands is located between the two recorder power-control units. When placed in the cross-coupled position by transmission of a digital auxiliary command, both recorders will play back sequentially in response to one ground-initiated playback command. For example, if a Playback 1 command is transmitted when the recorders are cross-coupled, the playback of tape recorder No. 1 will occur. After playback of tape recorder No. 1 has been completed, the cross-coupling circuit will automatically couple the playback power from system No. 1 to tape recorder No. 2. When cross-coupling is removed (by another digital auxiliary command) each tape recorder operates independently.

6. Camera-Clock Cross-Coupling

The two camera systems in the TV-picture subsystem contain identical units, and one of the two camera clocks in the programming and control subsystem is associated with each camera system. Cross-coupling of the digital clocks is provided to take advantage of this inherent redundancy, and simultaneous operation of both camera systems can be controlled by means of one digital clock when cross-coupling is commanded. When it is desired to operate each camera system independently, the cross-coupling for the camera clocks can be removed by ground command.

Operating power to a camera system is derived from the same power-switching circuit whether the system is operating in either the direct or the remote mode. When a Direct 1 tone is transmitted to the satellite, the power-switching circuit is switched on by an enabling level from command-control unit No. 1. When a remote sequence has been programmed, power is switched on when the programmed sequence starts. The power circuit for camera system No. 2 is identical to that of system No. 1, and cross-coupling of the camera clocks provides an additional means of switching either power circuit on.

The camera-clock cross-coupling relay (which is located in the command distribution unit) is connected between the two camera systems in a manner such that, when cross-coupling is programmed (by a ground-initiated digital auxiliary command) the camera-power output from one system is applied to the power-switching circuit of the other, causing the second to apply power to its camera system. For example, assuming that a Direct 1 command has been transmitted to the satellite, that the initial warm-up

period has expired, and that the programmer cross-coupling has been commanded, then the direct-1 enable level from command-control unit No. 1 is applied to turn on the power-switching circuits of system No. 1, camera No. 1 power is also applied through the closed contacts of the cross-coupling relay to the power-switching circuits that supply operating power to system No. 2.

It should be noted that the direct-picture requests initiated by momentary interruptions of the Direct 1 tone are simultaneously delivered to all four camera-triggering units. The activated unit on each side will, therefore, process the request and actuate the shutter of the related camera at the appropriate time. Since the cameras are located at diametrically opposite positions on the periphery of the spacecraft, the shutter actuations can occur within one-half spin revolution of each other, an interval of 3 to 5 seconds.

Cross-coupled operation in remote picture-taking sequences is identical to that in direct sequences, except that the initial power and picture requests are originated in one of the digital clocks. In either case, whenever one camera system is powered by command, the other will be powered through the cross-coupling relay.

Besides providing a major operating feature, the programmer cross-coupling increases system reliability through its inherent redundancy. Should a malfunction develop in one of the command chains in the programming and control subsystem, the operational capability of the satellite would not be affected, since both camera systems can be fully operated from one command chain.

7. TV-Transmitter Redundancy

The TV-picture subsystem includes a video coupler which provides a means of coupling the composite video subcarrier signals from both recorder systems to either of the two, identical TV transmitters. The subcarrier signals from both recorder systems are applied through the video coupler to the selected transmitter. Since the method of operation of the TV-picture subsystem prevents both subcarrier signals from occurring simultaneously, the output of only one camera system is coupled to the selected transmitter at any given time.

8. Camera Triggering

Four camera-triggering units (two computing triggers and two Dycons) are associated with the TV-picture subsystem. Each camera system uses one of the computing triggers as a primary means of shutter control, and a Dycon as a back-up. Both camera-triggering units associated with each camera system are supplied with the ground- or satellite-initiated picture request, but only one unit is energized on any single occasion.

The selected camera-triggering unit supplies the following signals necessary for camera-system operation· a run-tape command (in the case of the remote or direct-remote modes), vertical retrace pulses, and a shutter command. The run-tape command is used to start the tape recorder approximately 1 second before shutter actuation in order to permit the tape to reach operating speed. The vertical retrace pulses are sent to the camera for at least 6 seconds before shutter actuation to ensure that the vidicon is adequately prepared for exposure. The shutter command actuates the camera shutter and turns on the recorder signal electronics during the 2-second readout period.

If the Dycon is to be used for camera triggering, it is specified by ground command, and an inhibit signal is sent to the computing trigger. If the inhibit signal is missing and the picture power enable signal is present in the commanded camera system, the computing trigger will function as the camera triggering unit.

When the computing trigger is used, the unit computes the actual spin period and automatically determines the phasing of the camera vertical-sync pulses and the initiation of the run-tape command signal based on the computed spin-period value. The computing trigger also provides the PTDC signal, indicating in which of the two halves of the spin period the picture request occurred.

In the operation of the computing trigger, the unit receives horizon information from the associated horizon-scanner pair, through a threshold amplifier. As sky-to-earth transitions are sensed, the horizon scanners provide pulse outputs which are shaped by the threshold amplifier. The output of the threshold amplifier is applied to the associated computing trigger, where it is used to compute the correct time for shutter actuation.

Figure I-2 shows the satellite positional geometry during camera operation. As shown in this figure, each camera and its associated pair of horizon scanners are geometrically oriented with respect to each other such that, at the nominal design altitude, the horizon scanner is sensing the sky-to-earth transition when the camera is pointing at the local vertical. This orientation is slightly modified to accommodate system time delays such as the mechanical shutter delay. As shown in Figure I-3, the computing trigger is designed to operate with a specific spin-rate range, as determined by the altitude of the satellite.

When power is applied to the computing trigger, either during a direct or a remote picture-taking sequence, the unit begins timing the spin period of the satellite by counting out the interval between sky-to-earth transitions.

Since the computing trigger cannot differentiate between a sky-to-earth transition and an earth-to-sky transition, it performs an initial counter cycling during which it identifies the sky-to-earth transition. This is accomplished by inhibiting the horizon-scanner input after the initial pulse is received, for a period longer than the longest expected earth-time and shorter than the shortest expected sky-time, then locking on-to the next pulse -- the sky-earth pulse.

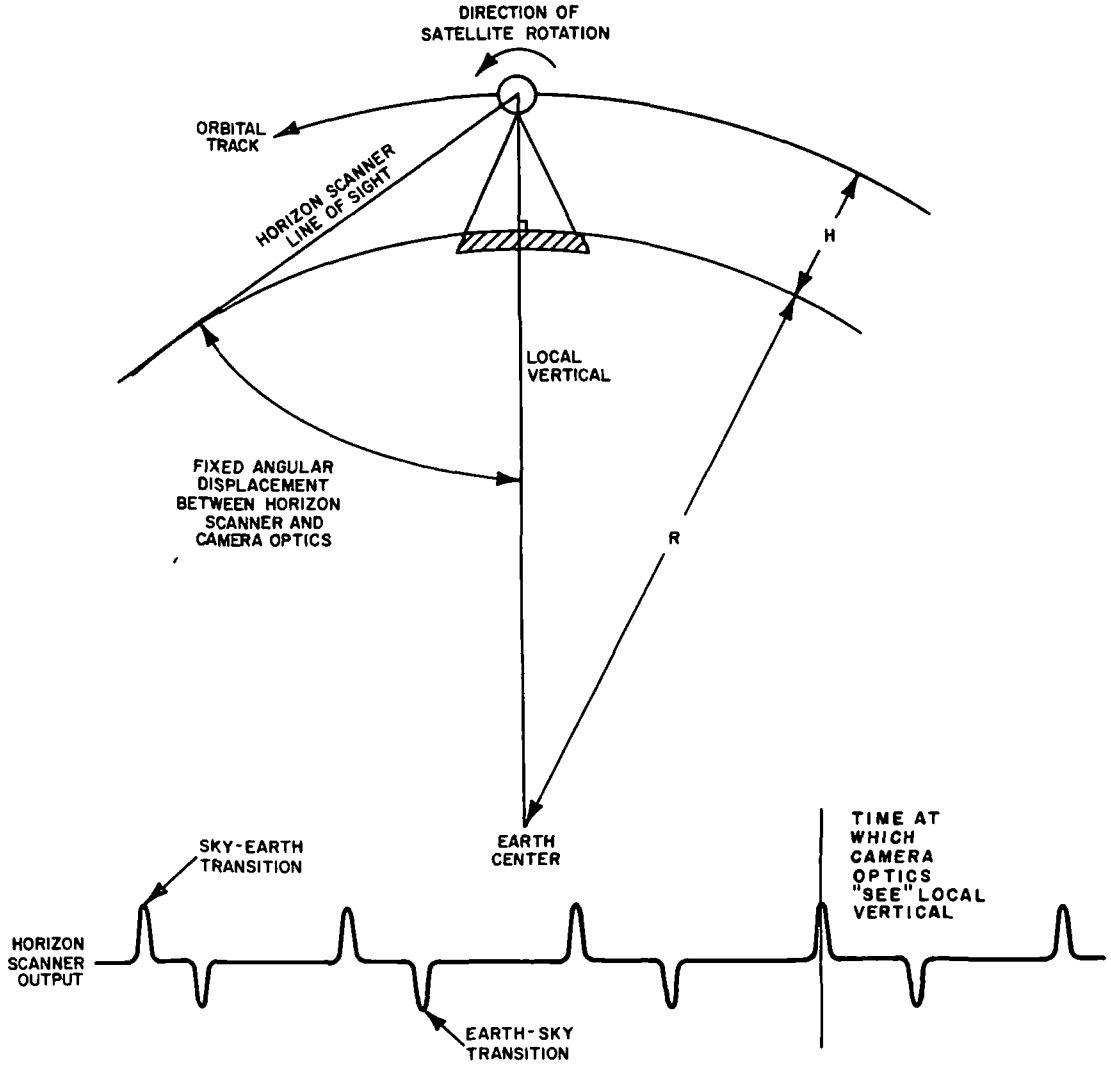


Figure I-2. Satellite Positional Geometry During Camera Operation

When a picture request has been sent to the computing trigger, the unit will not act on any further requests until the picture-taking cycle is completed.

When the Dycon functions as the camera trigger, the spin-synchronized principle of camera operation is used; in which the spin period is first determined, then adjusted by ground control, to be at or near 6.144 seconds. The camera-triggering function is then synchronized with this spin period, since 6.144 seconds is an integral multiple of the 2.048-second vertical-scan times.

The Dycon then synchronizes the vertical-sync pulses with the sky-earth pulses from the horizon scanners, effectively dividing each spin period into three 2.048-second intervals, with every third sync pulse coinciding with a sky-earth pulse. The shutter-actuation signal is then delivered coincident with a sky-earth pulse.

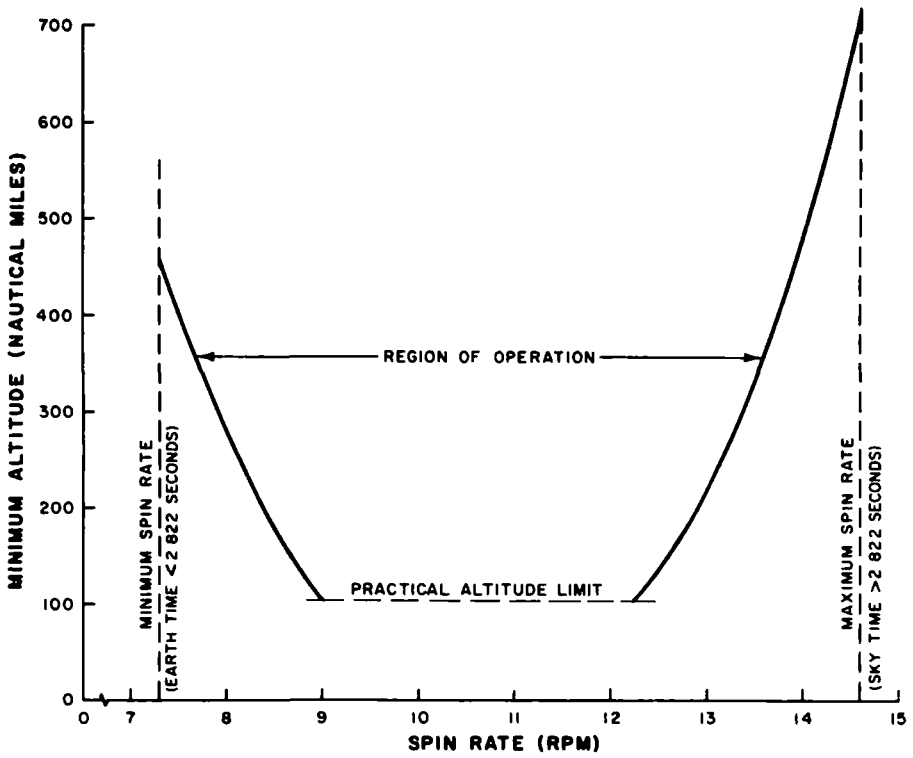


Figure I-3. Operational Range of Computing Trigger as Determined by Satellite Altitude

C SPIN-AXIS ORIENTATION AND SPIN-PERIOD DETERMINATION

As the satellite spins in orbit, the V-head attitude sensor and the horizon scanners yield an indication of the spin axis attitude with respect to the orbital plane, whereas the solar-aspect indicator yields an indication of the spin axis orientation with respect to the satellite-sun line.

The two infrared (IR) sensors in the attitude sensor are arranged in a "V" configuration, and, as the satellite spins, the optical axis of each head traces a conic section in space. The output of each head is supplied to a primary SCO for modulation on the beacon carriers. As noted, the attitude data is recorded at the CDA stations; and, from a comparison of the earth scan times of the two heads, the satellite's attitude can be determined. (In the perfect wheel attitude, the scan times are equal.)

The V-head attitude sensor is the basic system used to obtain an indication of spin-axis attitude, and this data is used in programming the operation of the QOMAC and MBC coils.

The four IR-sensing horizon scanners are arranged in two pairs, with the two scanners in each pair located at diametrically opposite positions on the baseplate. With this arrangement, one scanner in each pair is associated with each camera system, and the scanners are located on the baseplate such that when the scanners associated with a camera are sensing a sky-earth transition, the camera view axis is aligned with the local vertical.

One scanner pair is used consistently, while the second scanner pair is available as a redundancy feature.

During a direct-picture sequence, horizon-scanner data is applied to the secondary SCO on the commanded side and telemetered to the CDA station for the duration of the sequence.

The solar-aspect indicator yields a digital-coded measurement of the gamma angle, i.e., the angle between the spin vector and the satellite-sun vector when the sun is in the plane defined by the two vectors. Gamma-angle measurements are used in attitude studies and in power availability studies.

During initial direct or playback sequences, solar-aspect data is automatically obtained during the 28-second warm-up period of the command camera system, and is applied to the secondary SCO on the beacon transmitter on the commanded side for transmission to the CDA station. During this 28-second period, three or four measurements of the gamma angle will be obtained.

The time period between the leading edges on consecutive measurements of the gamma angle is an accurate measure of the satellite's spin period. The time period between consecutive sky-earth transitions in either channel of attitude data or in horizon-scanner data can also be used as the nominal spin period. The measurement of the spin period, as obtained from the solar-aspect indicator, is of sufficient accuracy for use in programming the operation of the MASC coil.

D. TRACKING AND TELEMETRY

The ESSA 1 tracking and telemetry operations are performed by means of a subsystem including two 50-mw beacon transmitters, four subcarrier oscillators (SCO's), two 90-point telemetry commutators, and a signal conditioner.

As the satellite proceeds in orbit, beacon signals are transmitted to the ground to aid the CDA stations in tracking and acquiring the satellite as it enters communications range.

Beacon transmitter No. 1 operates at 136.92 Mc; No. 2, at 136.23 Mc. Each beacon carrier is amplitude-modulated by means of two frequency-modulated SCO's to provide

a means of transmitting attitude and telemetry data to the CDA stations. The primary SCO on beacon transmitter No. 1 operates at 1300 cps, and the secondary SCO at 2300 cps. For beacon transmitter No. 2, the assignments are reversed. The primary SCO's with each beacon transmitter are continuously powered, and attitude data from the "up-looking" head of the V-head attitude sensor is continuously applied to the primary SCO on beacon No. 1, while data from the "down-looking" head is applied to the primary SCO on beacon No. 2.

Power is applied to the secondary SCO's when the satellite receives a command for a direct-picture sequence or a playback sequence, and a time-shared series of telemetry data is automatically applied to the secondary SCO's for transmission to the interrogating station. This data includes back-up attitude data, "housekeeping" telemetry data, solar-aspect data, horizon-scanner data, command-confirmation data, and MBC-switch stepping data. "Housekeeping" data can also be programmed by means of a specific command.

Two dual-channel paper-chart recorders are located at each CDA station for the recording of the data transmitted on the beacon signals. The attitude data transmitted by means of the primary SCO's is recorded on one recorder, while the other recorder is used for the secondary SCO's.

The "housekeeping" telemetry data provides an indication of the status of various satellite operating parameters. The telemetry signals are obtained in the form of voltage levels from sampling points within the satellite and are applied through the signal conditioner to each telemetry commutator, and thence to the associated secondary SCO.

The beacon transmitters can be temporarily turned off by ground command, at any time, for test purposes.

E. POWER REQUIREMENTS

During the course of the operational life of the satellite, the required electrical power is supplied by means of a solar-energy converter consisting of 9120 solar cells, and a battery pack consisting of 63 storage cells. During orbital day, the solar cells are used as the primary power source for the satellite's electrical system, while the battery pack is used during orbital night to supply all power required by the satellite. In cases where peak-power requirements during orbital day exceed the power output of the solar cells, the battery pack automatically supplies the power difference. A patch of 60 solar cells on the top of the spacecraft hat provides telemetry indications of the overall condition of the solar-cell array, permitting CDA station monitoring of any solar-cell degradation. Each camera system has its own regulated power supplies which operate at -24.5 volts and -13.0 volts.

F. ANTENNAS

A single dipole antenna is used by the satellite for the reception of commands from the CDA stations. For transmissions to the ground, two dual-frequency, crossed, dipole antennas and an RF matching and coupling network are utilized. A notch filter is also included in the antenna subsystem to ensure isolation of the command receivers from the beacon transmitters.

During a CDA-station interrogation, the RF matching and coupling network couples the three operating transmitters (one TV transmitter and both beacon transmitters) to the radiating elements, provides an impedance match for the coupling, minimizes interaction and feedback between the transmitters, and effects circular polarization by exciting the elements in phase quadrature.

G. RADIATION PROTECTION

A preliminary analysis of the possible effects of radiation damage on a wheel spacecraft in the ESSA 1 orbit was made prior to the TIROS IX program.* This analysis covered both the natural radiation environment (comprising the Van Allen radiation belts, galactic cosmic radiation, and solar-flare radiation) and the artificially produced radiation environment (the "Starfish" radiation belt). The total integrated electron and proton fluxes and their energy distribution were computed, using NASA-published data, as a function of the spacecraft orbital parameters, in order to determine the effects of charged-particle radiation on the degradation of spacecraft components and that of solar cells. NASA-published data was used also for determining the radiation-damage rate of the solar cells to be used on the spacecraft.

This analysis showed that the spacecraft design configuration could accept the anticipated radiation exposure and still perform the spacecraft OT-3 mission.

*A presentation of the radiation-damage analysis was contained in "Design-Study Report on THE TIROS-WHEEL SPACECRAFT WITH AN APT CAMERA, Astro-Electronics Division, Radio Corporation of America, Princeton, New Jersey, AED R-3026, 15 November 1963."

SECTION IV. PHOTOCOVERAGE FOR THE ESSA 1 MISSION

A DYNAMIC COVERAGE

Figure I-4 is a polar projection of the Northern Hemisphere, indicating the locations of the CDA ground stations and the minimum-elevation-angle contact circle for the antenna system at each station. The minimum elevation angle at the Wallops Island station is 5 degrees. Because of the antenna location at the Fairbanks station, the antenna pattern is such as to permit only a 10-degree minimum elevation angle in the direction due north and also southeast of the station.

Three successive orbit tracks are indicated on this projection. Orbit II represents the present orbit, Orbit I the preceding orbit, and Orbit III the next succeeding orbit. The sub-satellite point is noted at 2-minute intervals by a series of marks. The movement of the sub-satellite point from the 0-minute mark to the 24-minute mark traces an arc from the ascending node of the orbit; conversely, the movement of the point from the 24- to 48-minute marks traces the arc towards the descending node of the orbit.

In addition, three composite pictures, each in the characteristic "butterfly" pattern, are shown for Orbits I and II on this projection. The picture taken by the camera looking west, into the orbit track coming up, represents one wing of the "butterfly", while the other camera looking east, into the preceding orbit track, represents the other wing of the "butterfly". There is a 3-second time lapse (equivalent to 1/2 satellite spin period) between the two pictures representing the wings of the "butterfly" pattern. The rounded portion of the extreme edge of each wing of the "butterfly" pattern represents the earth horizon, and the small notch depicts where the picture represents the earth horizon, and the small notch depicts where the picture coverage does not reach the horizon but has a zenith angle of 80 degrees.

The picture overlap from orbit to orbit between pictures taken at the same latitude can be noted in this Figure. For purpose of graphic illustration, the interval between successive pictures on each orbit in Figure I-4 was made greater than normal. However, the in-track overlap between successive pictures can be visualized from the relative spacing of the 2-minute increments. If, for example, the picture interval had been set at 128 seconds, successive picture center points would almost fall on the 2-minute interval increments.

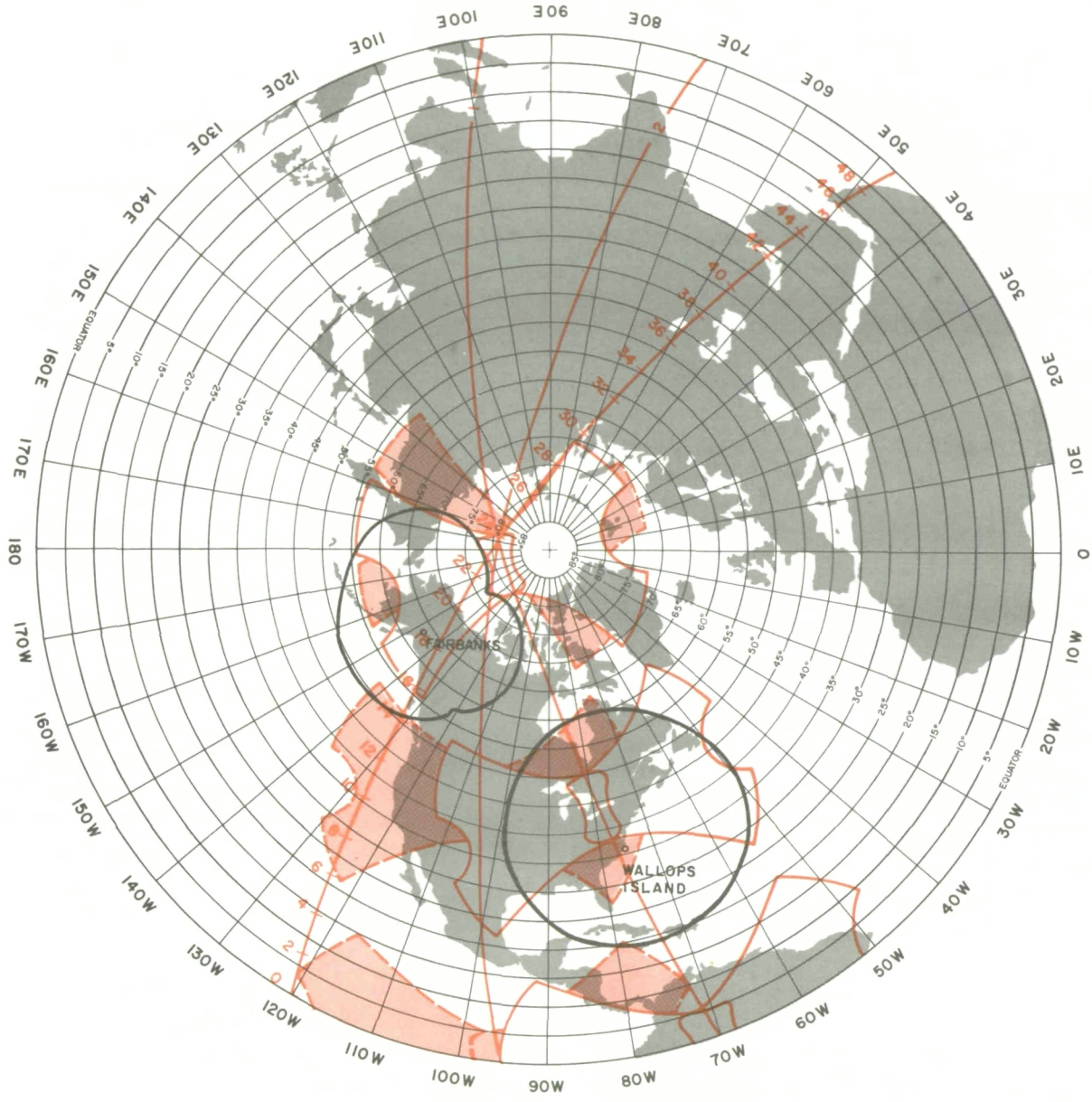


Figure I-4. Polar Projection of the Northern Hemisphere, Showing Subtrack for Three Consecutive Orbits of ESSA 1

To enable convenient visualizing of the orbit track of the satellite on the earth's surface (both Northern and Southern Hemisphere) with respect to the camera coverage and illumination boundries, a sub-satellite point ephemeris for three successive orbits at an altitude of 400 nautical miles has been superimposed on a Mercator projection of the earth and presented in Figure I-5. In this figure, the sub-satellite-point locations in 2-minute intervals are indicated by a series of marks; for reference, the two primary CDA stations and their respective contact circles at minimum elevation angles are also indicated.

B. ILLUMINATION

Because of the spherical shape of the earth, the solar elevation angle varies with latitude for a given time of year; and because of the tilt of the earth's axis with respect to the ecliptic plane, this variation changes as the earth moves around the sun. The choice of the time of the nodal crossing determines what variations of solar elevation angle will take place along the orbital track as well as the annual variations along the orbital track.

As noted earlier, a nodal crossing between 1400 hours and 1500 hours was chosen for the ESSA 1 mission. A comparison of the illumination characteristics available at either end on this time range shows higher illumination of the sub-satellite point resulting from the 1400-hours crossing. The even higher values available with a 1200 hours crossing are of limited advantage, however, because of extremely low satellite temperatures, low-energy output of the satellite solar-cell array, and the solar interference to the horizon scanners. Furthermore, the dynamic range of the camera system must be "adjusted" to compensate for a wider variation of light input. Since the dynamic range for the camera system is a constant, the only "adjustment" that can, in fact, be made is that of accentuating the higher brightness inputs to the camera at the expense of the lower brightness levels.

The more important consideration with regard to illumination is the total distance along the orbital subtrack over which the solar elevation angle is greater than 15 degrees (since 15 degrees is considered as the minimum solar elevation angle for obtaining useful pictures). This distance is not significantly different at the equinoxes or at the solstices for the 1400-hours and the 1500-hours nodal crossings. A high-noon orbit yields the greatest area in which the solar-elevation angles are above 15 degrees; however, it is a small increase as compared to the 1500-hours crossing.

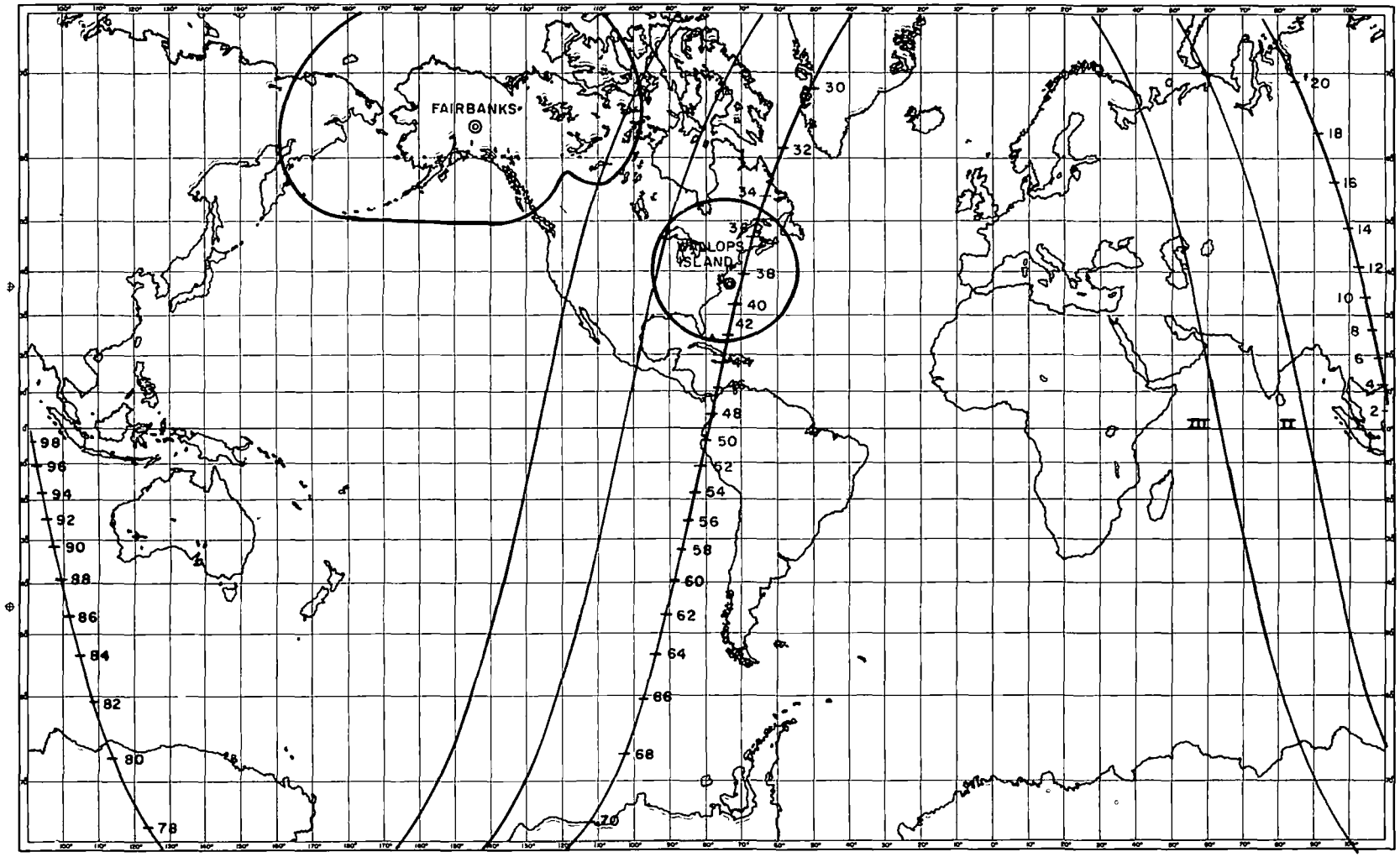


Figure I-5. Mercator Projection of World, Showing Three-Orbit Ephemeris for ESSA 1

The following observations can be made concerning the illumination boundaries produced with a 1500-hours crossing:

- There is satisfactory illumination of the Northern Hemisphere.
- There is excellent illumination of the "hurricane latitudes" in the hurricane season (i.e., between the summer solstice and the autumnal equinox).
- The ground sunlight time on each orbit varies only slightly over the period of a year.

From Figure I-4, it can be seen that the camera looking eastward into the previous orbit (the picture from which is represented by the right wing of the "butterfly" pattern) is looking into lower illumination levels than those at the sub-satellite point and will therefore see the illumination cutoff much more rapidly than the sub-satellite point. Secondly, the camera providing the westward-looking coverage, i.e., the left wing of the "butterfly" pattern, will be looking into brighter illumination levels than the sub-satellite point and will, therefore, continue to see acceptable illumination after the sub-satellite point has passed the illumination cut-off.

C. DETAIL COVERAGE

The projection of the camera field-of-view on the earth's surface for a 26.6 degree cant angle takes the form as shown in Figure I-6, because of the optical distortion of the lens system, the desired nadir angle, and the spherical shape of the earth. The origin of the graph is the local sub-satellite point (point B), and the line from points B to D indicates the direction of the orbit track for the satellite. The slight variations in back focal distance from lens to lens will alter the picture patterns that can be expected. For example, the Elgeet Navitar lens parameters used for the computation of this field-of-view projection were 72 degrees across the flats (side-to-side) and 104 degrees across the diagonals. If the computations were based on a lens with 74 degrees across the flats, the small notch which occurs in the edge of the projection would disappear and only a smooth horizon would be evident. This variation in back focal distance is less than 3 percent and is not uncommon. In addition, the variation in the respective alignments of the camera optics, fudicial references, and displacements of electrical centering, with camera environment and life, will also affect the amount of horizon contained in the resulting picture. At the angles near the horizon, the tangent function becomes quite large and a small difference in lens coverage will have a very significant affect on horizon coverage.

The field-of-view of the lens system used in the ESSA-1 cameras covers a large portion of the earth since the cameras are canted from the local vertical, and there is quality resolution coverage of an area of approximately 360,000 square miles. This area of quality resolution is shown in Figure I-6 as the darkened area around the picture center and represents picture elements for which the zenith angle is less than 60 degrees. The effective resolution in the quality coverage area is better than 2 nautical miles per TV line.

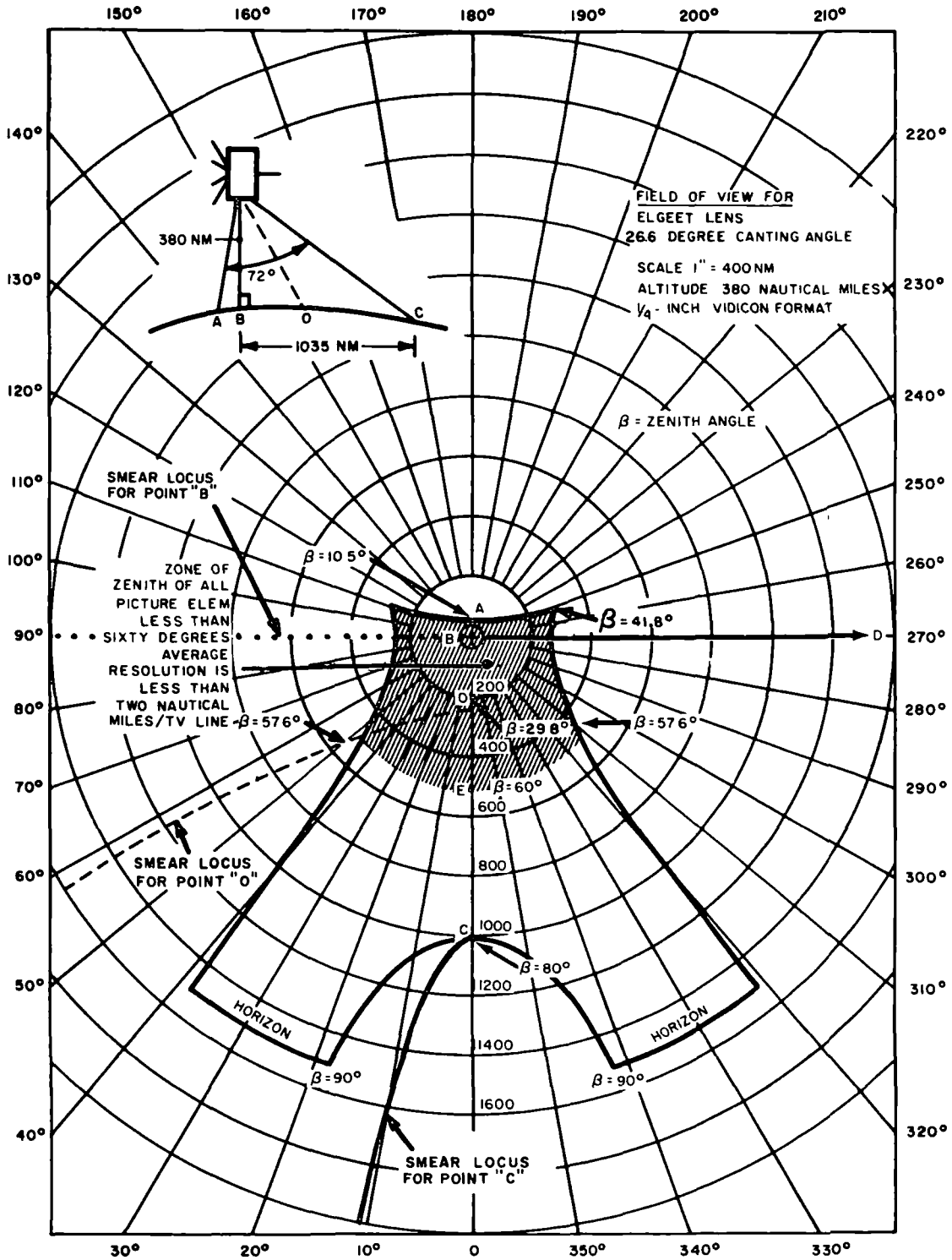


Figure I-6. Field-of-View, for a 400-Nautical-Mile Orbit, of a Standard TIROS Wide-Angle Camera

The horizon is visible along one edge of the resultant picture. The resolution of the 500-scan-line, 0.5-inch, standard TIROS vidicon camera is approximately 375 lines in the horizontal direction and 375 lines in the vertical direction. There are approximately 318 effective resolution lines between points A and E, and only 57 resolution lines between points E and C. Between points E and C, where the zenith angles range from 60 to 80 degrees, the resolution is a rapidly degrading function as a result of the small number of resolution lines and the tangential effect near the horizon. Figure I-7 is a translation of the field-of-view of the earth's surface as shown in Figure I-6 into the 1/4-inch vidicon format. The points A, B, C, D, and E in each figure and the 60-degree zenith-angle circles are identical. The shaded portion in Figure I-7 represents the horizon.

The effective picture smear was also shown in Figure I-6. The ground-track locus of the sub-satellite point, picture center, and picture edge are plotted on the graph. If the camera system is operated before the system reaches the minimum nadir angle, the sub-satellite point (point B), the picture center (point O) and the picture edge (point C) will lie along the smear locus curves as shown. During the time that the camera shutter is in motion (1.5 milliseconds) the points B, O, and C move along the

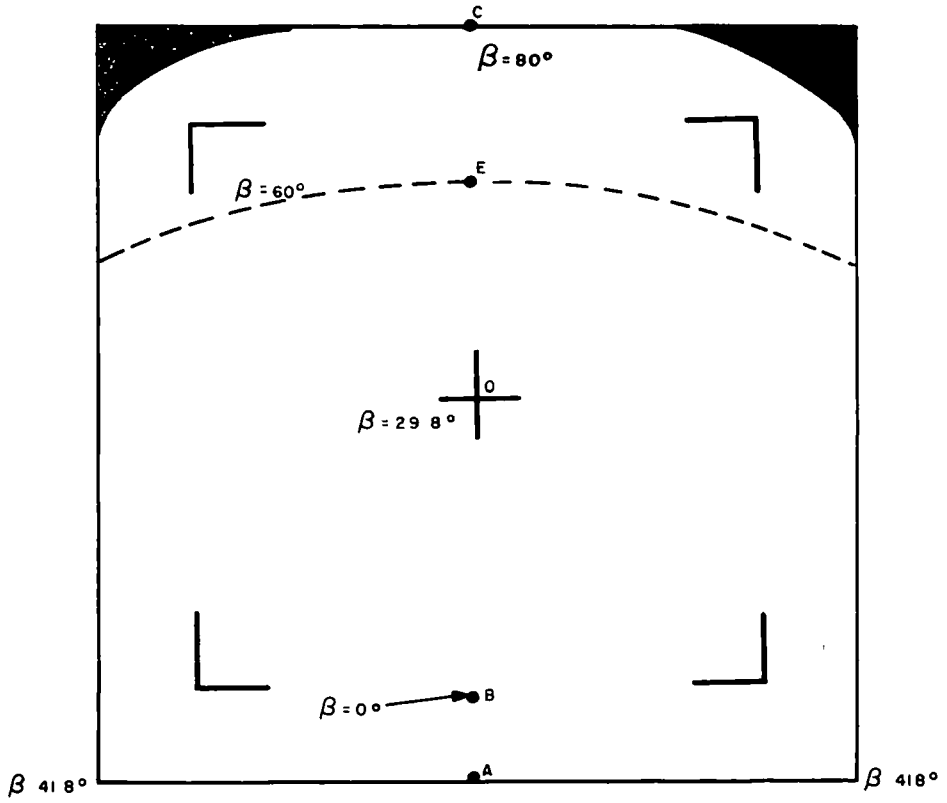


Figure 1-7. Coverage with Elgeet Lens at 400-Nautical-Mile Altitude with Camera Canted at 26.5 Degrees

locus curves and cause picture smear. If the satellite spin rate is assumed to be a maximum of 12 rpm, the angular rotation of the satellite during the exposure time is as follows:

$$\theta = 12 \frac{Rev}{Min} \times \frac{360^\circ}{Rev} \times \frac{Min}{60\ Sec} \times 0\ 0015\ Sec = 0\ 108^\circ$$

The ground distance, i.e., the in-track motion, at point B during the exposure time is as follows:

$$X_B = (0\ 108^\circ) \times \frac{Rad}{57\ 3^\circ} \times 400\ NM = 0\ 80\ NM$$

which is considered to be negligible.

The ground distances for points O and C during the exposure time will vary along the locus curves but will be nearly equal to that for point B in the region of the minimum nadir-angle pictures associated with the camera-triggering operations for a wheel spacecraft. If the pictures were taken when the camera optical axis was substantially away from the minimum nadir location, the resulting smear of points O and C would be degraded by the rapid motion of the points along the loci.

The coverage overlap between successive orbits can be best illustrated by examining the worst case overlap conditions which occur along the circumference of a great circle which lies in a plane normal to the orbit plane. Such a plane, whose inclination to the equatorial plane would be 8 degrees, is shown in Figure I-8. Projections of the camera coverage are illustrated along the circumference for the fifteen orbits of the day. The overlap between successive orbits, the gap between alternate orbits, and the overlap between orbits 1 and 15 are illustrated in this Figure.

D. PICTURE ORIENTATION AND RECTIFICATION

Orientation of the pictures received from ESSA-1 can be readily accomplished, since the camera optical axis maintains a fixed angular displacement with respect to the orbit plane, which is at a fixed inclination to the earth's poles.

Location of the picture is derived from the determination of the satellite's position in orbit at the time of picture taking and the camera nadir angle.

The time of initiation of a remote-picture sequence, the duration of the interval between picture requests, and the camera nadir angle will be known. However, actual

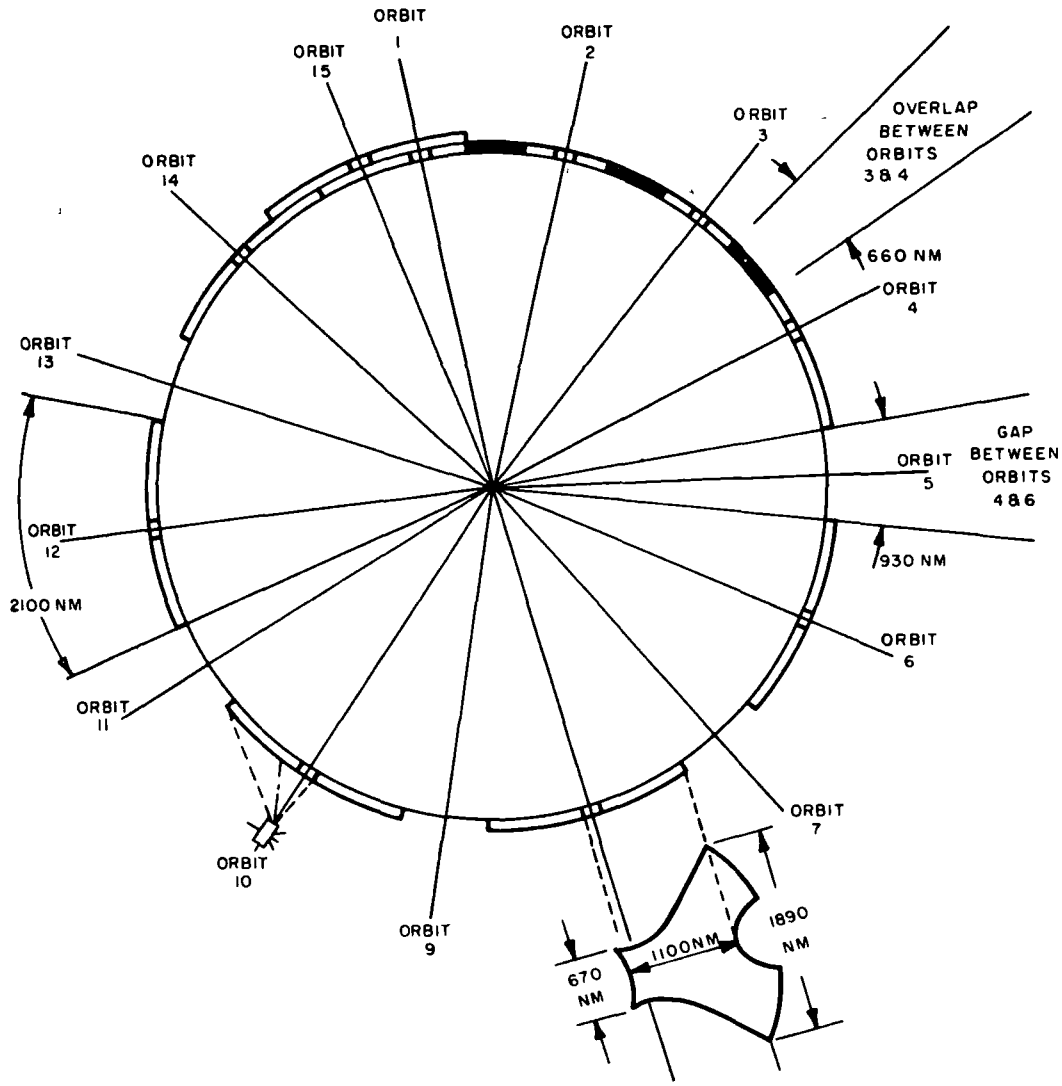


Figure I-8. Picture Coverage With A Standard TIROS Wide-Angle Camera in the Plane Normal to the Orbital Plane, at a 400-Nautical Mile Orbit

shutter operation does not occur at the instant of picture request. Instead, the camera-triggering unit witholds shutter initiation while calculating the time at which the camera optical axis will reach the minimum nadir angle. For a minimum nadir angle, the shutter is triggered at the time of a specified sky-earth horizon pulse following the picture request; the actual time of the shutter pulse is referenced to the occurrence of a sky-earth horizon pulse the horizon scanner. Thus, the only time interval that will not be known directly will be the interval between the picture request and the next sky-earth horizon pulse.

The time interval (T) between the picture request and the next sky-earth horizon pulse can be determined in the following manner. Since the satellite spin-rate during an orbit is essentially constant and the exact Greenwich Mean Time (GMT) of a sky-earth pulse can be determined at the time of programming, the times of all subsequent sky-earth pulses occurring during the picture sequence can be predicted. Since the alarm time and the interval between picture requests are known, the time of any specific picture request can be predicted and the time interval (T) between a horizon pulse and a picture request can be determined. Thus, the picture-request time and "T" can be used to give the GMT of the picture taking. Using the known satellite position at this GMT, the satellite altitude, and the camera nadir angle, the picture center can be located and orientation accomplished.

PART II. SPACECRAFT DESIGN AND TESTING

SECTION I. DESIGN OF SPACECRAFT COMPONENTS

A. INTRODUCTION

The ESSA 1 satellite, i.e., the TIROS spacecraft OT-3, is almost identical in design, layout and structure to TIROS IX, the first wheel-type meteorological satellite. The overall block diagram of spacecraft OT-3 is presented in Figure II-1. The subsystem breakdown of the spacecraft is basically similar to that of each of the ten TIROS satellites which preceded spacecraft OT-3, and is as follows:

- the TV-picture subsystem (which consists of two identical camera systems utilizing standard TIROS TV cameras),
- the programming and control subsystem,
- the telemetry and tracking subsystem,
- the reference-indicator subsystem,
- the power-supply subsystem,
- the antenna subsystem, and
- the dynamics-control subsystem.

The various modifications made to the spacecraft OT-3 subsystems for the ESSA 1 mission are detailed in the following in the respective subsystem descriptions. The most significant of these were those made to the TV transmitters and magnetic tape recorders.

The physical dimensions of spacecraft OT-3 are identical to those of the previous TIROS satellites. The spacecraft is an 18-sided polyhedron with a height of 22.5 inches and a diameter of 42 inches.

B TV-PICTURE SUBSYSTEM

1. General

The TV-picture subsystem in ESSA 1 consists of two identical TV-camera chains, camera systems Nos. 1 and 2, each of which contains a 0.5-inch vidicon TV camera,

a magnetic tape recorder capable of storing a minimum of 48 TV pictures, and a TV transmitter. In addition a video coupler is included for application of video data from either or both camera systems to either TV transmitter. Figure II-2 is a functional block diagram of the TV-picture subsystem. As shown in the diagram, each camera system has, as associated equipment, the following units:

- A camera clock which controls remote-picture sequences.
- A computing trigger which functions as the primary means of camera triggering.
- A Dycon unit which functions as a back-up means of camera triggering.
- A pair of horizon scanners which function as a reference system in both methods of camera triggering.

The ESSA-1 camera systems are capable of operation in either the direct, remote, direct-remote, or playback modes. Detailed descriptions of camera-system operation in each of these modes are presented in Part I, Section IV of this report (System Operation).

The computing triggers and the Dycon units are logically components of the programming and control subsystem, which is described in Volume II (the classified supplement) of this report. However, since the computing triggers are so closely involved in camera operation, operational and functional descriptions of these units are included herein with the description of the TV-picture subsystem.

The operation of the Dycon units in camera triggering is also covered in this description. However, greater detail on the Dycon units is presented in the description of the dynamics-control subsystem, since the Dycons are also closely related in the operation of the QOMAC and MASC coils.

2. TV-Picture Subsystem Components

a. Television Cameras

(1) Equipment Description

The TV cameras employed in ESSA 1 utilize a ruggedized, 1/2-inch vidicon developed for the TIROS I program. A 500-line-per-frame scan is used, and readout is accomplished at a line rate of 250 cps (non-interlaced) and a frame rate of 0.5 cps (2 seconds per frame). Vidicon exposure is implemented by a focal-plane

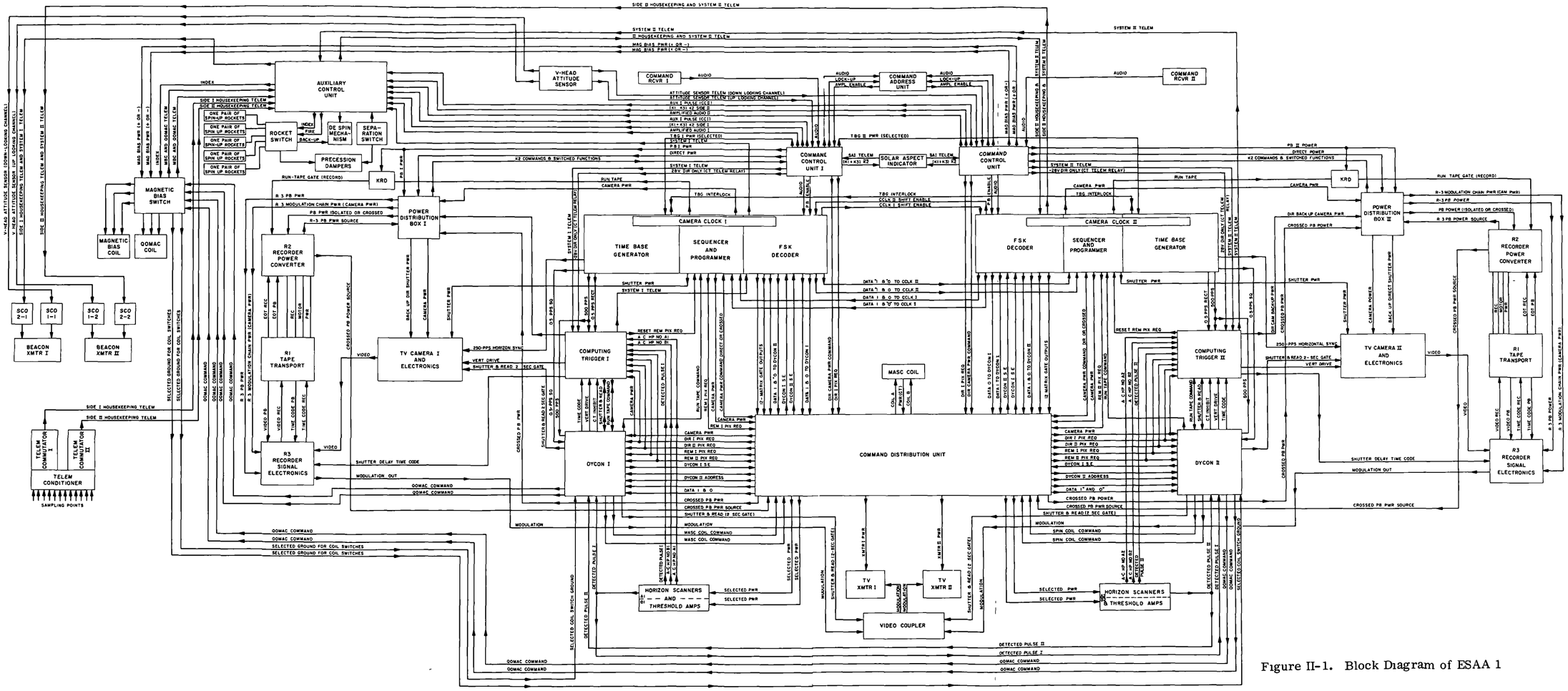


Figure II-1. Block Diagram of ESAA 1

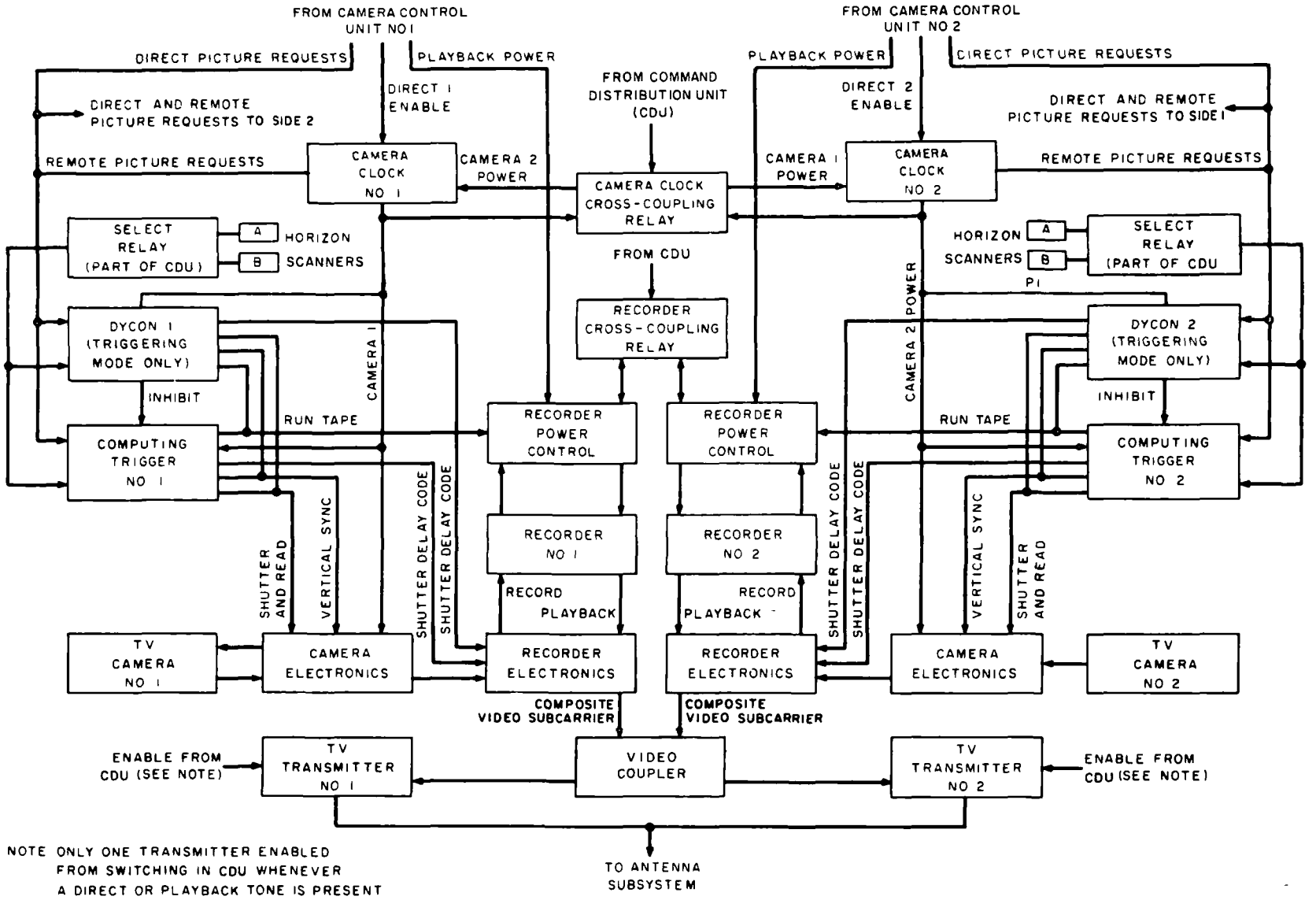


Figure II-2. TV-Picture Subsystem, Functional Block Diagram

shutter, and the exposure time is 1.5 milliseconds. The combined weight of the TV camera and associated electronics is approximately 9.5 pounds.

(2) Functional Description

Figure II-3 is a block diagram of the TV camera and camera-electronics unit. The camera lens focuses the light gathered from the viewed scene onto the photoconductor of the 1/2-inch vidicon. The photoconductor has a scanned area of 1/4-inch square. Reticle lines are superimposed on the photoconductor surface, including a cross at the center and corner markings which define a 3/16-inch square. The reticle markings are used for optical-mechanical calibration and linearity determination.

The camera uses a focal-plane, traveling-slit shutter which is driven by a moving coil in a magnetic field. The shutter blade is accelerated quickly, travels across the face of the vidicon at nearly constant speed, and is stopped at the end of travel by a silicone-rubber bumper. The exposure time is nominally 1.5 milliseconds. Power to the shutter coil is supplied by switching the coil terminals between -26 volts and ground with power transistors. The terminal voltages are reversed for each operation, causing the shutter to move from left to right on one operation, and then right to left on the succeeding operation. Shutter actuation is synchronized by the horizontal sync pulses and occurs at the end of the vertical blanking period. The exposure is completed before the start of the vertical sweep (i.e., at the end of vertical blanking). A horizontal deflection waveform is generated at a rate of 250 pulses per second. The sawtooth-deflection waveform is amplified and applied to the horizontal-deflection yoke. Symmetry (centering) and amplitude (size) controls are included for initial camera adjustment.

A vertical-deflection sawtooth waveform is generated at a rate of 0.5 pulse-per-second. The amplified vertical-deflection voltage is applied to the vertical winding of the deflection yoke. Centering and size controls are included to permit adjustment of the scanned area to accommodate the individual vidicon being used.

The vidicon power supply contains a regulated dc-to-dc converter to provide electrode voltages and a current regulator for the focus-coil and filament currents. The supplied voltages are minus 20 and minus 30 volts for the control grid, 20 to 30 volts for the target, 250 volts for electrostatic focus, and 300 volts for grid No. 2. A current regulator supplies 100 milliamperes, ±1 percent, for the magnetic-focus coil and the vidicon heater, which are connected in series. The remaining bias voltages and preamplifier heater and plate voltages are obtained directly from the satellite's solar-conversion power supply.

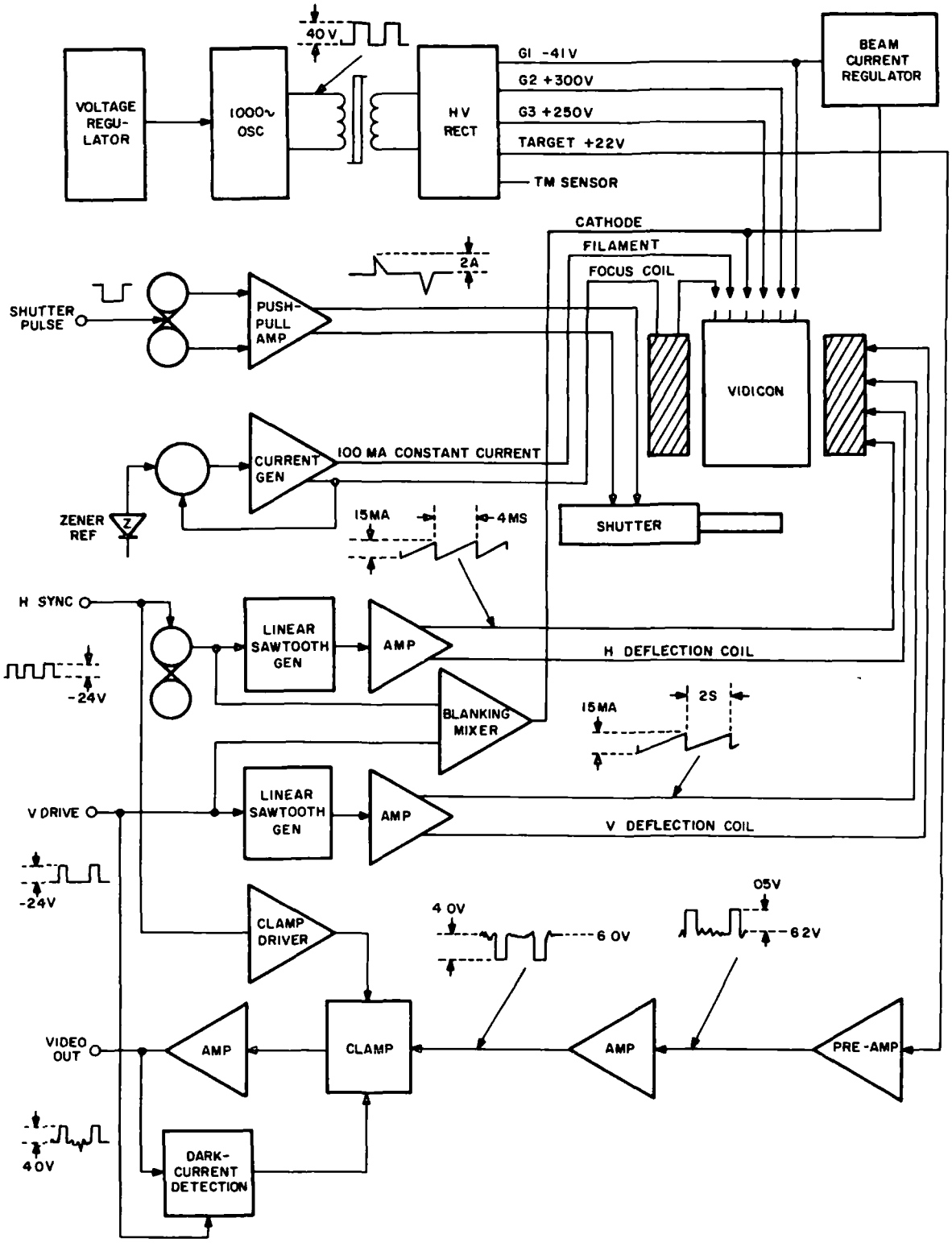


Figure II-3. TV Camera And Camera-Electronics Unit, Block Diagram

The beam-current regulator is included to ensure that the intensity of the scanning beam remains relatively constant and is unaffected by factors such as vidicon aging, or ion contamination. The regulator performs its function by changing the voltage applied to the grid G1 whenever there is a tendency for the beam current to change.

b. Tape Recorders

(1) Equipment Description

Each of the ESSA 1 magnetic tape recorders consists of a tape transport (R1), a signal-electronics unit (R-3), and a power-electronics unit (R2). The sole purpose of the power electronics is to provide operating power to the tape transport during record and playback sequences. The tape transport and the signal-electronics unit function as an integral component during record and playback, and comprise two separate data processing and storage channels. One of these channels is for video information and the second is for the Picture Time-Delay Code (PTDC). In addition. the signal-electronics unit is activated during direct operation to process and mix the video and PTDC data, and route the resulting complex signal to the TV transmitter.

The power requirements for the signal electronics unit are (1) in direct operation: 24.5 vdc, 45 ma, (2) in playback operation: 24.5 vdc, 10 ma, and (3) in record operation: 24.5 vdc, 40 ma. The power requirement for the power electronics unit in both playback operation: 26 vdc, 0.9 ampere, and (2) in record operation: 26 vdc, 0.9 ampere. (The power-electronics unit is not used when the camera system is operating in the direct mode.)

The tape transport consists of two coaxial reels, a capstan, and an idler mounted in a triangular configuration. The capstan and idler are angled to guide the tape as it moves from the plane of one reel to that of the other. A constant-tension spring assembly between the reels maintains uniform tape tension regardless of the amount of tape on each reel. The capstan is belt-driven by a synchronous hysteresis motor. Recording is performed by means of RCA-VR502 heads, and playback by means of RCA-VR402 heads. A permanent magnet provides the erase capability.

The tape transport contains a polyester tape 830 feet long, 0.0011 inch thick, and 0.375 inch wide, and operates at a tape speed of 50 inches per second. End-of-tape sensors are located adjacent to the record and playback heads to sense when the end-of-tape position is being reached on the emptying reel. The sensors are actuated by silvered leaders, 4 feet in length, attached near each end of the magnetic tape. As back-up to these sensors, each reel is equipped with a lever-operated microswitch,

as shown in Figure II-4. The microswitches sense the "full" condition of the take-up reel at a specific level of tape build-up. The lever is adjusted such that this level occurs 4 to 8 turns after the point at which the end-of-tape sensor is scheduled to operate.

Both the end-of-tape sensors and the microswitches operate to remove power from the tape transport before the end-of-tape position has been reached. As further back-up to these devices, each reel is also equipped with a teflon bumper which is located adjacent to the microswitch and which exerts pressure on the tape to stop the reels from turning. The bumper is set to "operate" 4 to 6 turns after the scheduled operation of the associated microswitch. (The bumpers are not shown in Figure II-4.)

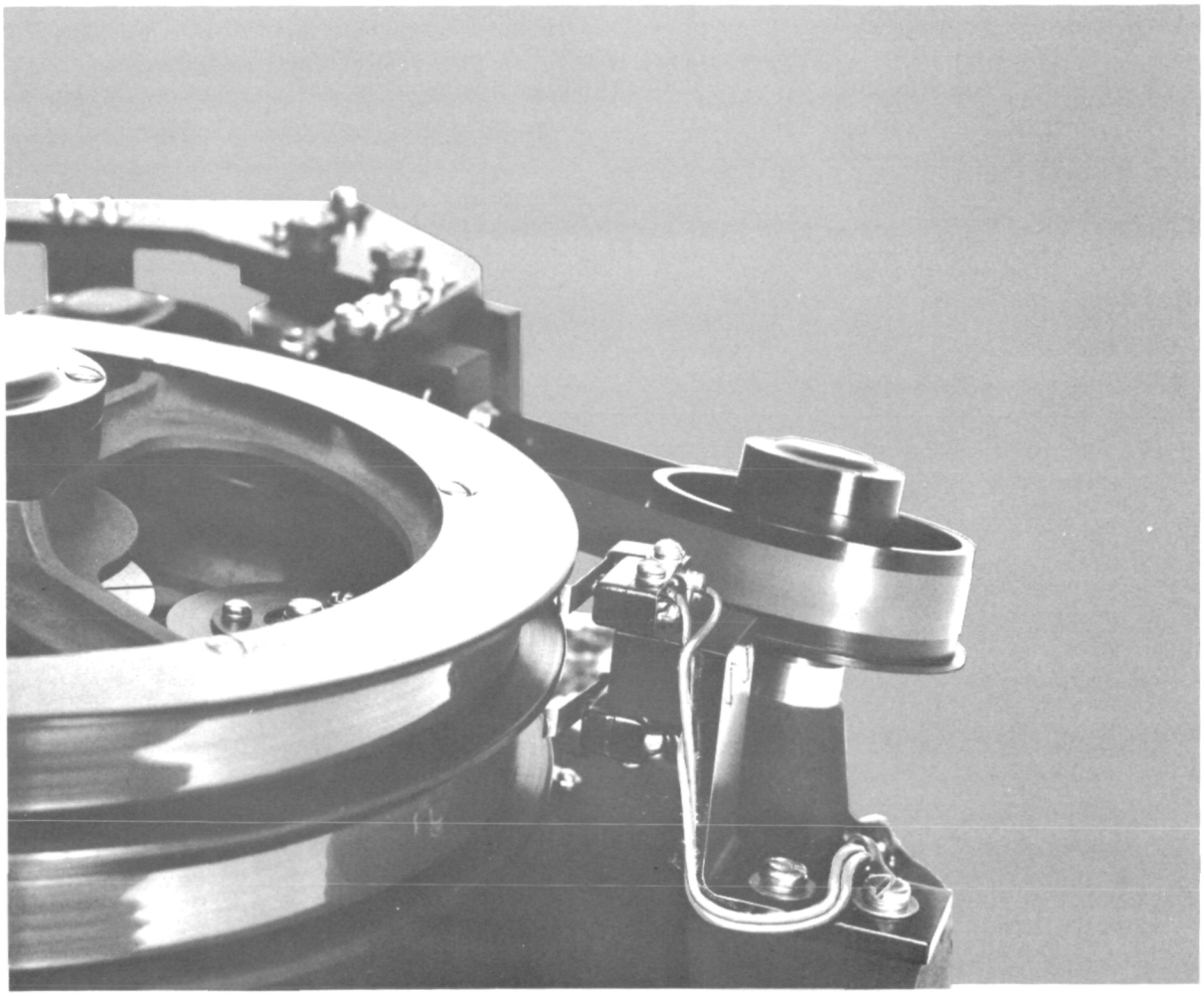


Figure II-4. Tape Transport

(2) Functional Description

(a) Direct Mode

Figure II-5 is a block diagram of the ESSA 1 tape recorder. During direct operation, the signal-electronics unit is enabled by operating power from the TV camera and command-control unit for the selected camera system. Power from the TV camera is applied to the 10-kc gated oscillator, emitter follower, and fast rise-time circuit. The fast rise-time circuit is enabled during the 2-second readout period and applies operating power to the modulator and head drivers. Power from the command-control unit is applied to the mixer and amplifier-limiter.

Composite video signals from the camera are applied through an emitter follower to the video modulator circuit, where the amplitude variations of the video signal produce frequency variations of an 85-kc subcarrier oscillator. The resulting frequency-modulated signal is applied, in turn, to the mixer. The serial, binary-coded pulses representing the Picture Time-Delay Code (PTDC) from the camera trigger gate the 10-kc gated oscillator. The output from the oscillator is in the form of bursts of 10-kc oscillations having durations equal to corresponding pulses of the PTDC input, and is applied to the mixer along with the video signals. After mixing, the composite video-PTDC signal is applied through the amplifier-limiter to frequency-modulate the TV transmitter.

(b) Record Mode

During the record cycle, i.e., remote operation, picture regulated power from the TV camera is applied to the 10-kc oscillator, emitter follower, and, during each readout period, through the fast rise-time circuit to the modulator and head drivers. Also during remote operation, unregulated power is applied to the power-control circuit of the power-electronics unit. This power, which is controlled by the camera-triggering unit, is applied 1 second before each shutter command and is maintained until vidicon readout is complete.

Nominally 1 second before the vidicon shutter is actuated, unregulated power is applied through the power-control circuit of the power-electronics unit to the 500-cps converter. This converter is synchronized by the 500-pps output of the time-base generator (in the camera clock) and supplies a 500-cps, 110-volt squarewave through the motor-reversing relay to the synchronous motor of the tape transport. Because the motor-reversing relay is de-energized during record, the squarewave applied to the motor windings causes the tape transport to move in the record direction.

The composite video resulting from vidicon readout is applied through the emitter follower to the modulator and produces frequency variations in

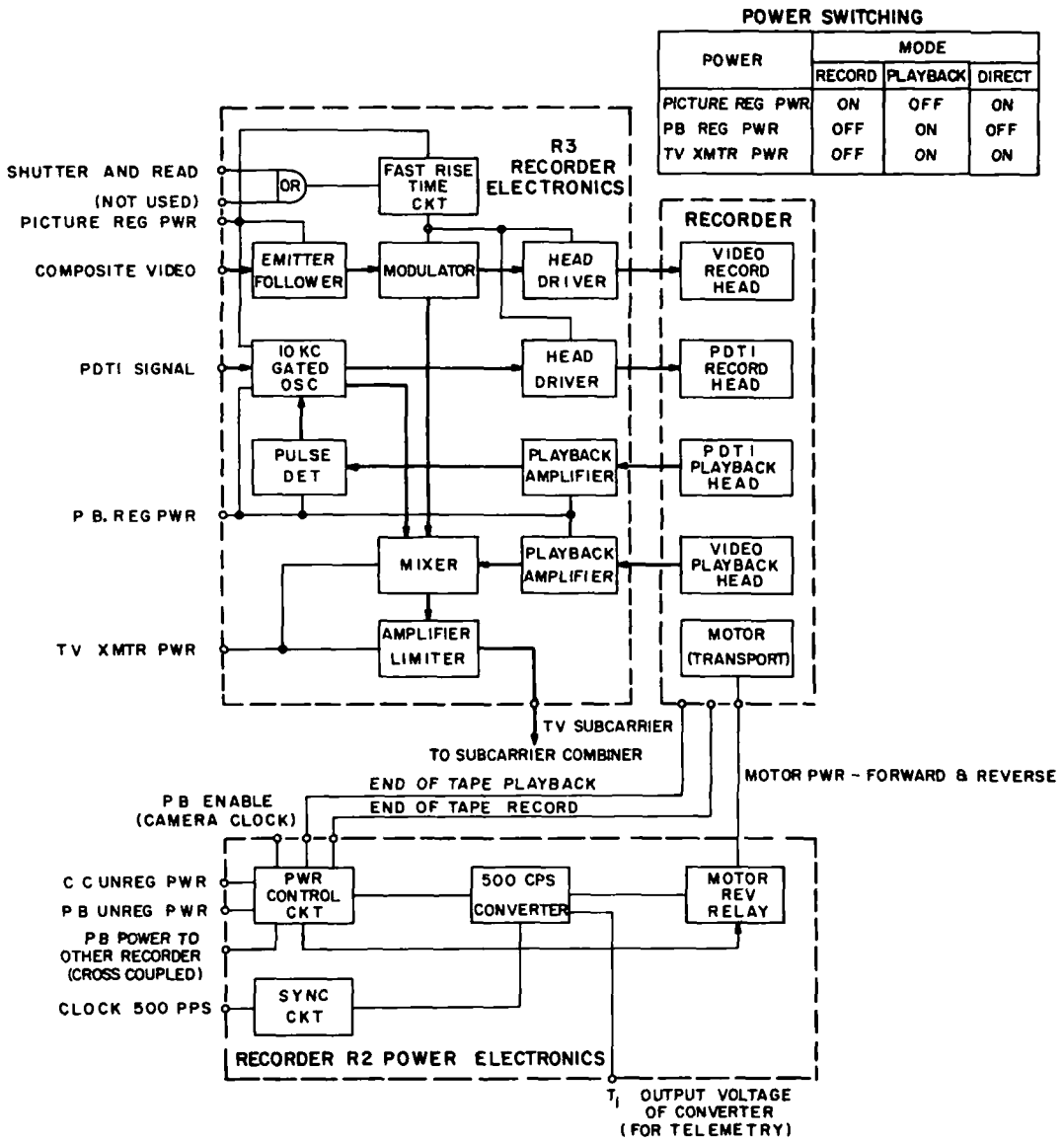


Figure II-5. Tape Recorder, Block Diagram

the 85-kc subcarrier oscillator. This FM signal is then applied through the head driver to the video record head of the tape recorder.

The PTDC signal from the camera trigger gates the 10-kc oscillator and, thus, produces a series of bursts of 10-kc oscillations. These bursts are applied through the head to the PTDC record head of the tape transport.

At the end of the record period, the fast rise-time circuit is disabled and, in turn, disables the modulators and head drivers. Also, unregulated power is removed from the power-electronics unit, causing the tape transport to halt operation until 1 second before the next shutter trigger. When the overall remote sequence ends, regulated power is removed from the signal-electronics unit, causing the tape transport to stop. When the overall remote sequence ends, regulated power is removed from the signal-electronics unit, the recorder power control is switched to playback and all three components of the tape recorder remain disabled until the start of playback.

(c) Playback Mode

During the playback mode of operation, operating power is applied to the tape recorder from the command-control unit or from the power electronics of the other tape recorder when in the cross-coupled mode of operation. Playback unregulated power is applied to the power-electronics unit and enables the 500-cps converter and energizes the motor-reversing relay. With the relay energized, the 500-cps signal from the converter is coupled to the motor winding and causes the motor to run in the playback direction.

The data recorded on the video channel is read out by the video-playback head and amplified in the playback amplifier before being applied to the mixer. The 10-kc bursts from the channel 2 (PTDC) playback head are amplified, detected, and used to gate the 10-kc oscillator. The "re-gated" 10-kc bursts are summed with the video signal in the mixer. The resulting composite signal is amplified by the amplifier-limiter and is applied through the video coupler to frequency-modulate the TV transmitter.

When tape playback is complete, an end-of-tape signal is sent from the tape transport to the power-electronics unit. This switches the recorder to the record state and interrupts playback power to the motor, causing the tape transport to stop. Because the tape is erased as it is played back, the tape recorder is ready for the next remote sequence.

c. TV Transmitters

The two TV transmitters used in ESSA 1 are identical 5-watt units which operate in the 235-Mc telemetry band. Each transmitter has a video bandwidth greater than 200 kc and is capable of low intermodulation at 150-kc deviation. Figure II-6 shows a block diagram of the TV transmitter. A temperature-stabilized, voltage-controlled oscillator, Q1, operates at about 21 Mc and is frequency-modulated by variable-capacitance diode PC-117. Buffer amplifier Q2 feeds mixer Q3 where the output of crystal oscillator Q4 is added to produce an output frequency of 78.333 Mc. This output is then frequency-tripled by Q6 and amplified by Q7, Q8, and Q9 to drive the final amplifier Q10-Q11. The nominal power output of the transmitter is 6 to 8 watts, with a minimum of 5 watts under environmental extremes. Regulator Q12 controls the bias for amplifier Q10-Q11 to minimize the effect of drive and temperature variations.

d. Video Coupler

An interconnection and block diagram of the video coupler is shown in Figure II-7. The video coupler provides a means of coupling the composite video subcarrier signals from both camera systems to the input of either TV transmitter.

The composite video-subcarrier signals for a camera system are generated in the tape recorder signal-electronics (R3) unit. The enable signal from the signal-electronics unit is gated ON or coupled to the TV transmitter input only at the times when the video subcarrier signals are also present at the output of the signal electronics. Since the operation of the camera systems prevents both subcarrier signals from occurring simultaneously, only one signal-electronics unit will be coupled to the transmitter input at any given time. By controlling the power to the transmitter, the selected transmitter will be used to transmit either or both composite video subcarriers as they are applied through the video coupler to the input of the transmitter.

3. Camera-Triggering Units

a. Computing Trigger

Each camera system has an associated computing trigger which functions as the camera-triggering unit when (1) the picture power enable signal is present and (2) at the same time, the inhibit signal from the associated Dycon is absent. The computing trigger measures the spin period from horizon-scanner information and automatically determines the phasing of the camera vertical-sync pulses and the initiation of the run-tape command signal, based on the computed spin-period value.

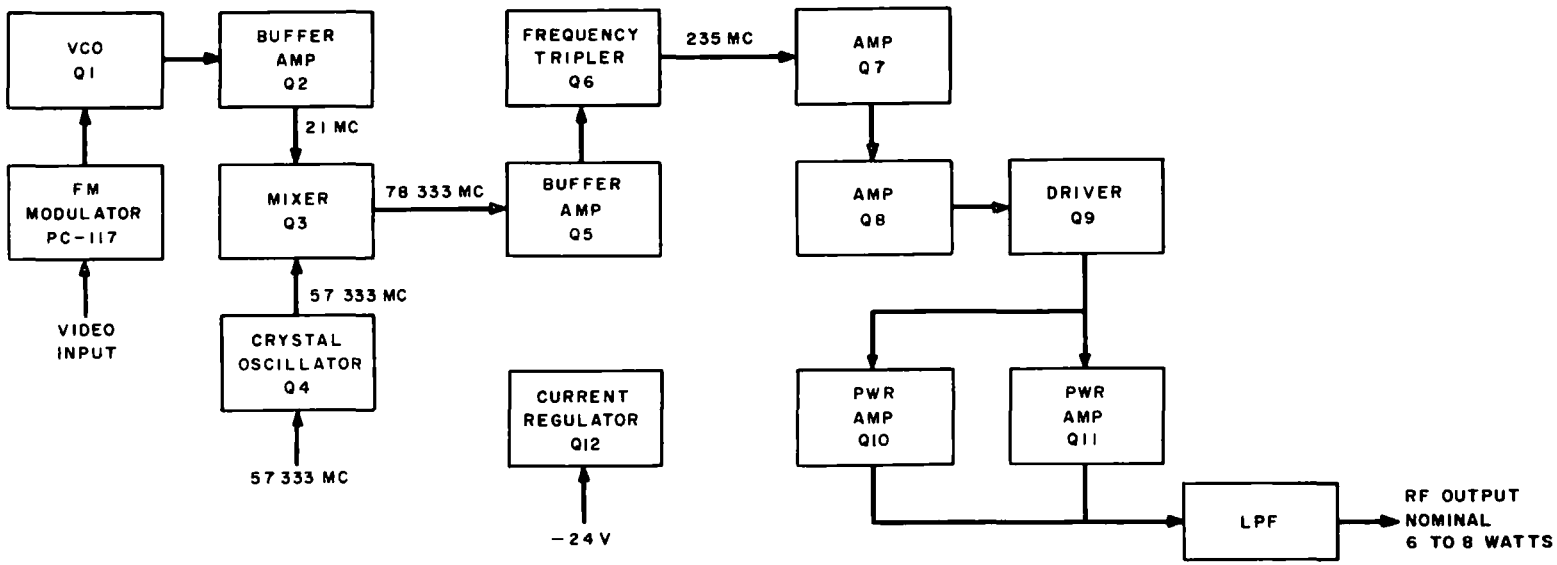


Figure II-6. TV Transmitter, Block Diagram

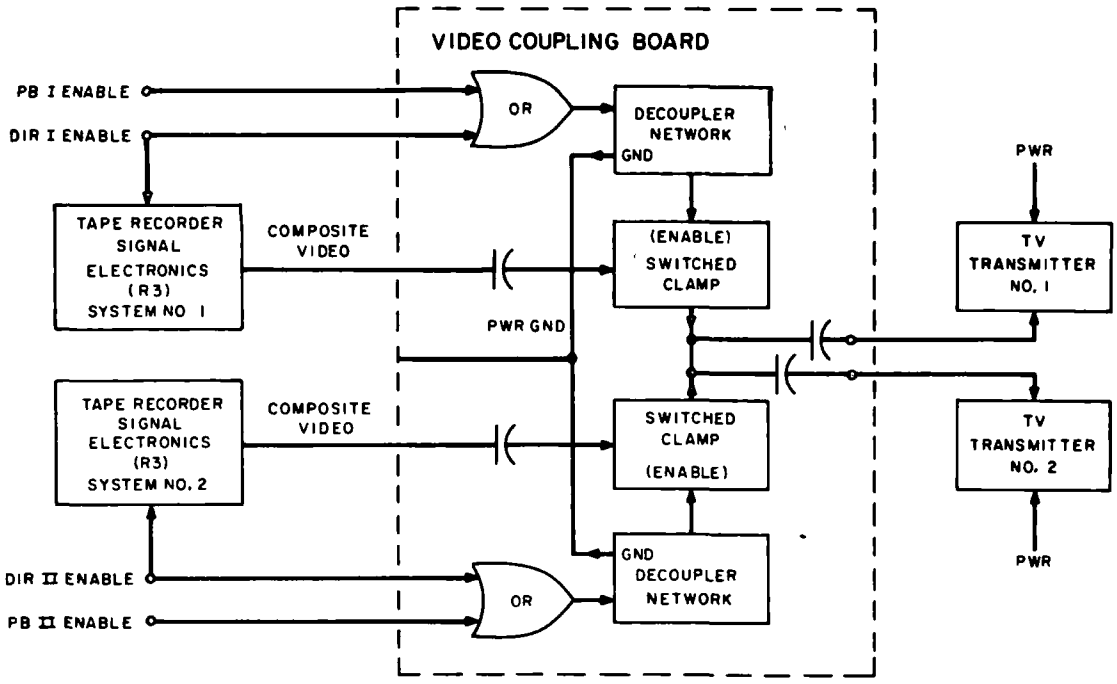


Figure II-7. Video Coupler, Interconnection and Block Diagram

Figure II-8 shows a timing diagram of the computing-trigger operation. On the first sky-earth pulse after the picture request has been received, the computing trigger, using spin-period data acquired on the previous spin revolution (i.e., the revolution in which the picture request occurred), begins computing the time at which shutter actuation will occur.

At 8.192 seconds before the anticipated shutter actuation, the computing trigger begins sending vertical sync and blanking pulses to the camera at 2.048-second intervals. These pulses, which initiate and time the erase and prepare operations on the camera vidicon, are so synchronized that the fifth blanking pulse will coincide with the third sky-earth pulse, and, therefore, initiate the camera readout. If the horizon pulse which causes the shutter-actuation signal occurs earlier than predicted, the last 2.048-second interval is shortened. If the horizon pulse occurs later than predicted, the blanking portion of the vertical drive signal that occurs at the end of the 8.192-second interval is lengthened. The horizon pulse can be as late as 128 milliseconds before the logic circuits initiate a shutter-actuation signal.

A run-tape command is delivered to the tape transport 0.86 second before the third sky-earth pulse, thereby allowing sufficient time for the tape transport to achieve operating speed. When the third sky-earth pulse is received, the shutter-actuation command is sent to the camera. Vidicon read-out occurs in the following 2-second interval, after which the run-tape command is removed. The computing trigger is then ready to receive another picture request.

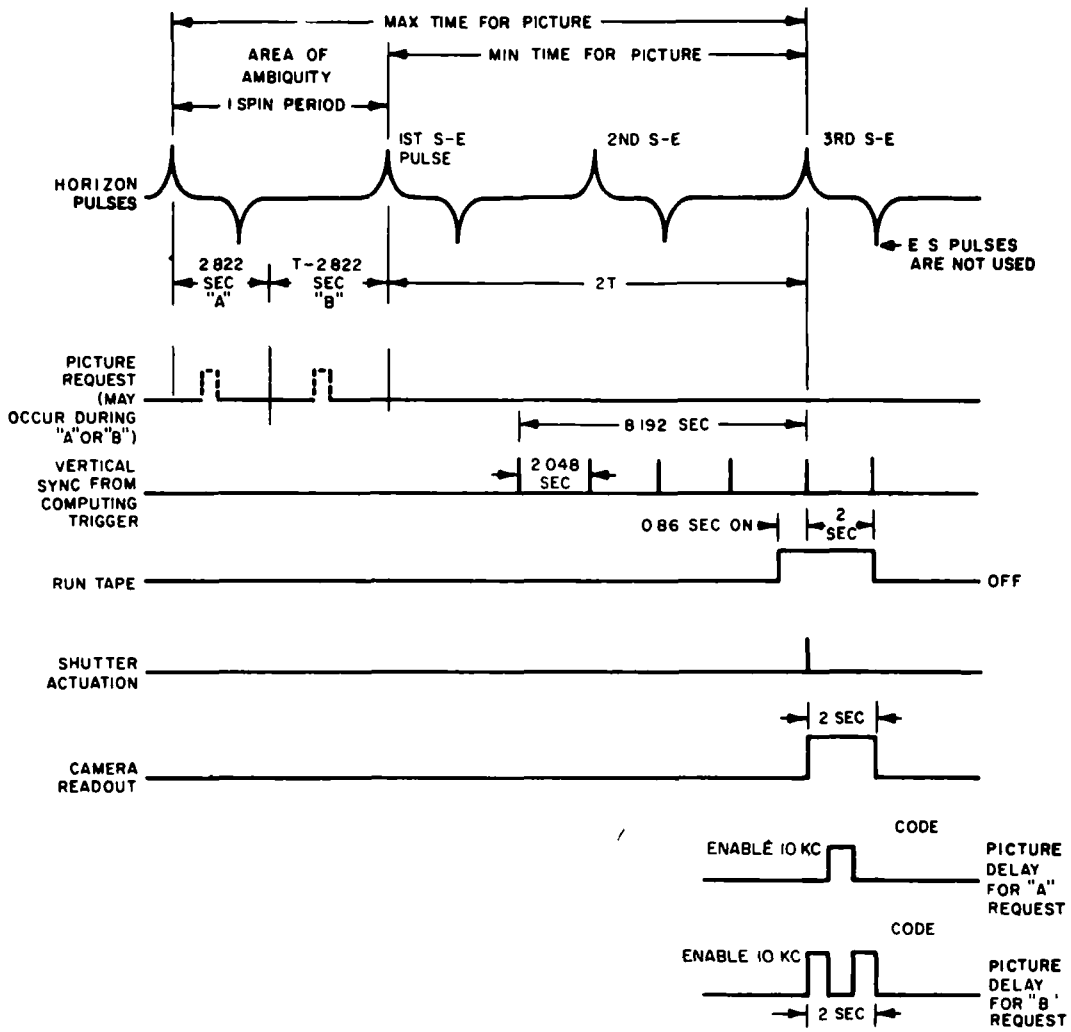


Figure II-8. Timing Diagram, Operation of the Computing Trigger

It should be noted that the minimum period required to process a picture request is longer than two spin periods but shorter than three. This presents an ambiguity, on the order of 1 spin period, in determining the time elapsed between the picture request and shutter actuation, i.e., the picture-delay period. This ambiguity presents no problems during direct-picture operations, since other means are available at the ground stations to determine the exact shutter time. However, in remote operations, an ambiguity of this order is not acceptable. In order that the real time of the shutter actuation can be determined more accurately during remote operation (so that the geographical center of the picture can be determined more accurately during meteorological interpretation), the computing trigger generates a reference signal indicative of picture-delay time, and records this signal, in the form of 10-kc bursts, on one channel of the tape recorder.

The Picture Time Delay Code (PTDC) indicates in which of the two portions of the spin period the picture request occurred, i.e., (1) whether the picture request occurred during the first 2.822 seconds following a sky-earth transition, or (2) whether the picture request occurred during the remainder of the spin period. Since the operational spin period of the satellite is approximately 6 seconds, this reduces the ambiguity by nearly half.

A block diagram of the computing trigger is shown in Figure II-9. The computing trigger consists of a 12-stage forward/backward counting register which is controlled by a sequencer. The sequencer is operated by information from a horizon-scanner pair. During a picture-taking sequence, the sequence controls the counter which, in turn, is decoded at specific counting levels to change the state of the sequencer. The counter functions at two different rates: namely; full rate in the forward direction, and half rate in the backward direction. The counter by-passes flip-flop RA1 to count forward at a 500-pps rate and includes RA1 to count backward at a 250-pps rate. Power is switched to reduce the standby requirement, and picture requests are restored by the computing trigger, which will execute the first request received before starting a new computation. The sequencer controls the interface circuits to the outputs.

The computing trigger performs a computation automatically when camera power is applied in the absence of an inhibit signal from the Dycon, and pictures can be taken at minimum intervals of 22.5 seconds at the minimum spin rate.

b Dycon Unit

The Dycon unit* associated with a camera system functions as the camera-triggering unit when specified by ground command (causing an inhibit signal to be sent automatically to the associated computing trigger). In order to permit use of the Dycon in camera triggering, the spin period of the satellite is first brought to, and held constant at (or near) 6.144 seconds, which is an integral multiple of the 2.048-second vertical-scan times. When the Dycon has been commanded and energized, the vertical-sync pulses are synchronized with the sky-earth transition pulses from the associated horizon scanner. Each spin period is, therefore, divided into three, 2.048-second intervals, with every third sync pulse coinciding with a sky-earth pulse. Figure II-10 is a timing diagram of Dycon operation in the camera-triggering mode.

If a picture request is received by the Dycon within the first 5.120 seconds following a sky-earth pulse (picture-request A, Figure II-10), the camera shutter will be

*Further detail on the Dycon unit, along with a schematic diagram, is presented in Section II-H.

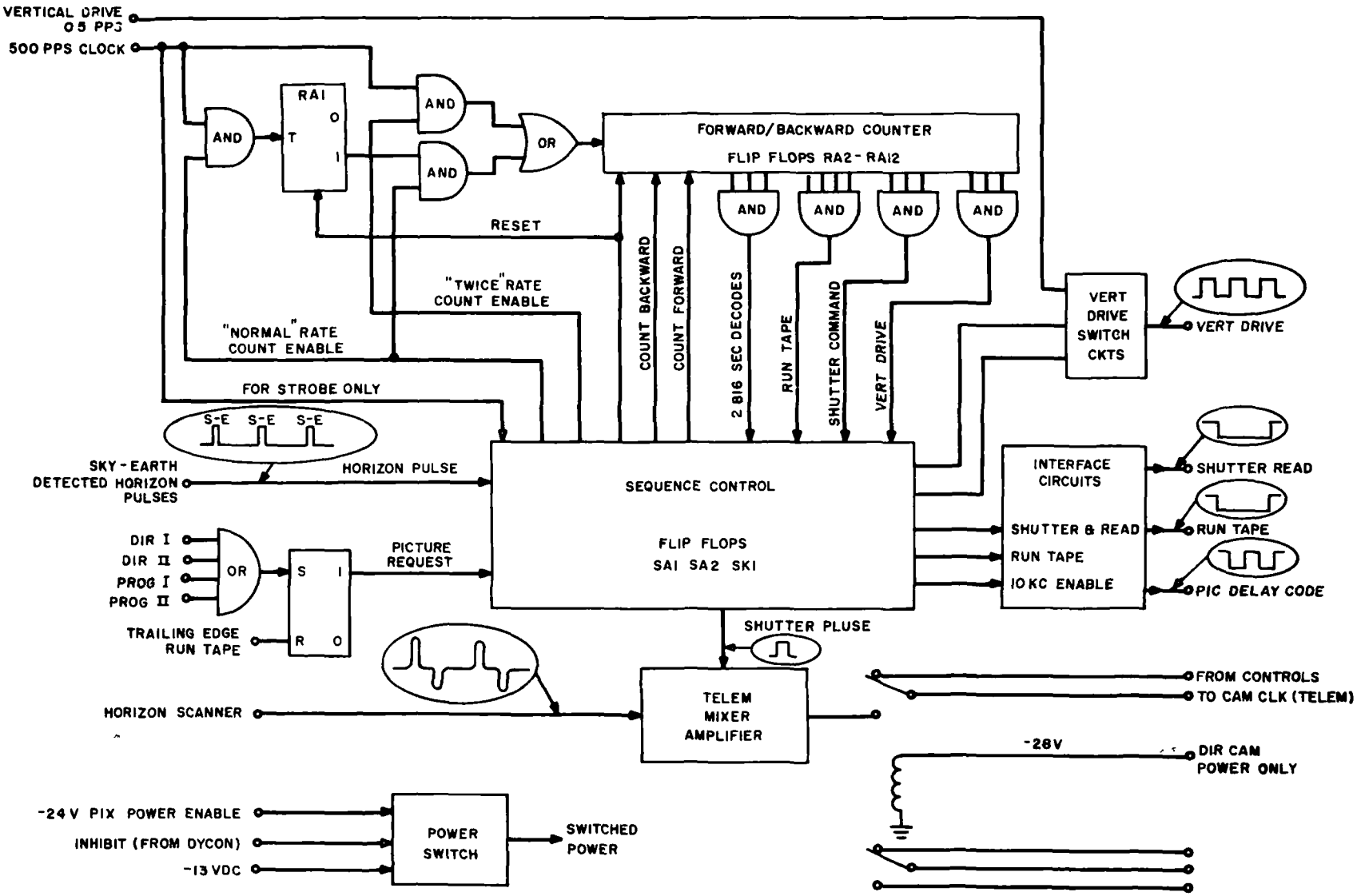


Figure II-9. Computing Trigger, Block Diagram

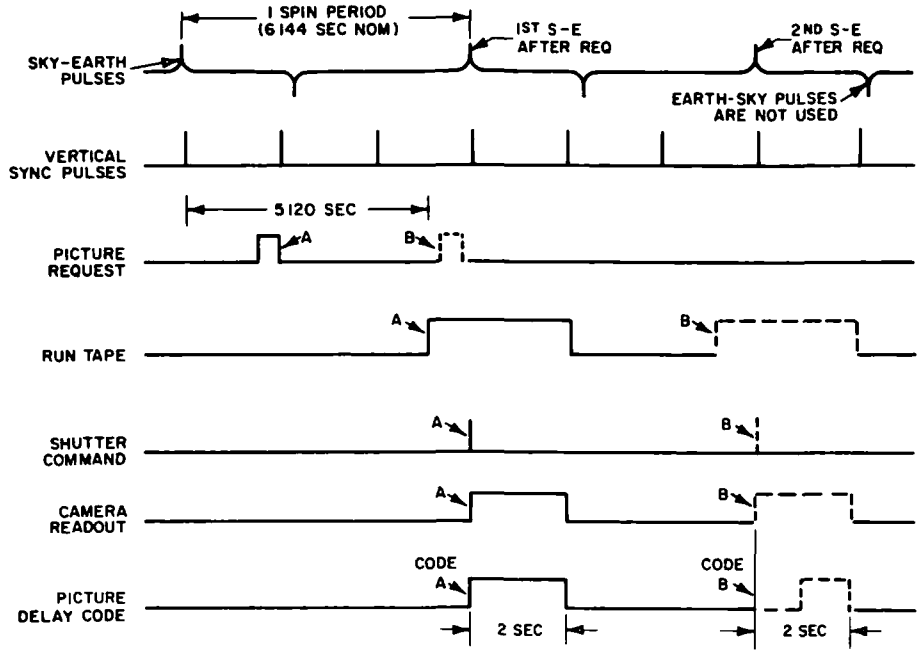


Figure II-10. Timing Diagram, Operation of the Dycon Unit in Camera Triggering

actuated upon receipt of the next sky-earth pulse, when the associated camera is again looking downward. As soon as camera power is applied, the spin-synchronized vertical sync pulses are delivered to the camera for the erase and prepare operation and, approximately 1 second before the next sky-earth pulse is received, the run-tape command is initiated. The shutter-actuation signal and the vertical-sync pulse are delivered coincident with the sky-earth pulse.

If the picture request is received after the 5.120-second period following a sky-earth pulse (picture-request B, Figure II-10), the shutter will not be actuated on the following sky-earth pulse, which would provide insufficient time for the tape transport to reach synchronous speed. Instead, the shutter will be actuated on the second sky-earth pulse following the request. The remainder of the operation, however, is identical to that with the computing trigger.

4 Testing

a. TV Camera No 1 and Camera Electronics Unit No 1

The vidicon of TV camera No. 1, Serial No. 3-221-08, was optically aligned to the camera lens system, using an optical alignment bench, on March 2, 1964. During target alignment checks on July 2, 1964, the optical axis of the camera lens system and

the fiducial center of the vidicon were found to be misaligned by 2.5 degrees. The alignment was readjusted on July 7, and calibration photographs were obtained which indicated the alignment to be within 0.5 degree.

Vibration testing of TV camera No. 1 and its associated electronics unit, Serial No. 3-222-08, was completed with satisfactory results on March 3, 1964; thermal-vacuum testing was successfully completed on March 14, 1964.

Further vibration testing of camera electronics unit No. 1 was performed on August 20, 1964, after diode CR 2, TK-21, had been replaced with a USN-IN 3189 diode. (This replacement was made because of a problem encountered in the qualification testing of TIROS IX, further detail on the modification can be obtained from the TIROS IX Final Engineering Report.)

Prior to delivery for integration, the spectral response of each TV camera was measured to determine the effects of the non-visible light spectrum on the vidicon output. The tests were made using a Perkin-Elmer 112-U spectrometer and a monochrometer. The signal from the camera was amplified in the video circuits and measured as a video voltage on an oscilloscope. Figure II-11 and II-12 are plots of the spectral response of camera Nos. 1 and 2, respectively.

Because of a microphonics condition observed in November 1964 during operation of the cameras as part of the debugging effort on spacecraft OT-3, the shutter assembly on each camera was replaced. Shutter Serial No. 13 was installed on TV camera No. 1 on November 12, 1964.

b. TV Camera No. 2 and Camera Electronics Unit No. 2

The vidicon of TV camera No. 2, Serial No. 3-221-11, was optically aligned to the camera lens system, using the optical alignment bench, on April 17, 1964. Target alignment checks were performed from August 21 to September 1. During these checks slight misalignment between the optical axis of the camera lens system and the fiducial center of the vidicon was noted. However, adjustment of the optical alignment corrected the condition.

TV camera No. 2 and its associated electronics unit, Serial No. 3-222-11, were successfully subjected to vibration testing on April 21, 1964. Thermal-vacuum testing was initiated on April 23, but was halted after 2 days, with the temperature level at -10° C, when erratic exposure times became apparent. After the shutter had been replaced, testing was resumed; thermal-vacuum testing of TV camera No. 2 and electronics unit No. 2 was successfully completed on May 2, 1964.

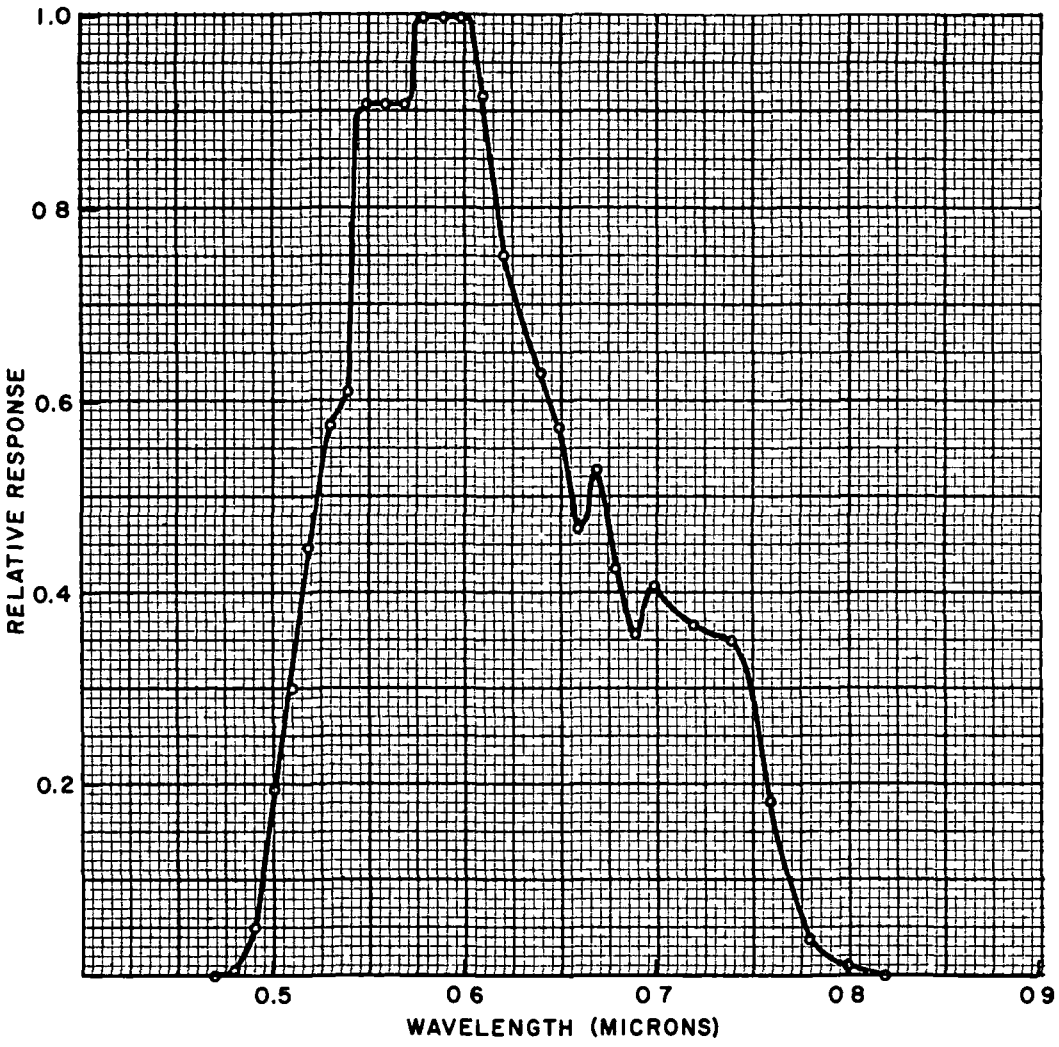


Figure II-11. Camera No. 1, Relative Spectral Response

The replacement of the TK-21 diode in camera electronics unit No. 2 with a USN-IN 3189 diode was performed in August 1964, and the required additional vibration testing of the electronics unit (in the thrust axis only) was performed on August 21 with satisfactory results.

As noted earlier, the shutter assemblies on the spacecraft OT-3 cameras were replaced during the debugging effort, shutter Serial No. 02 was installed on TV camera No. 2 on November 13, 1964.

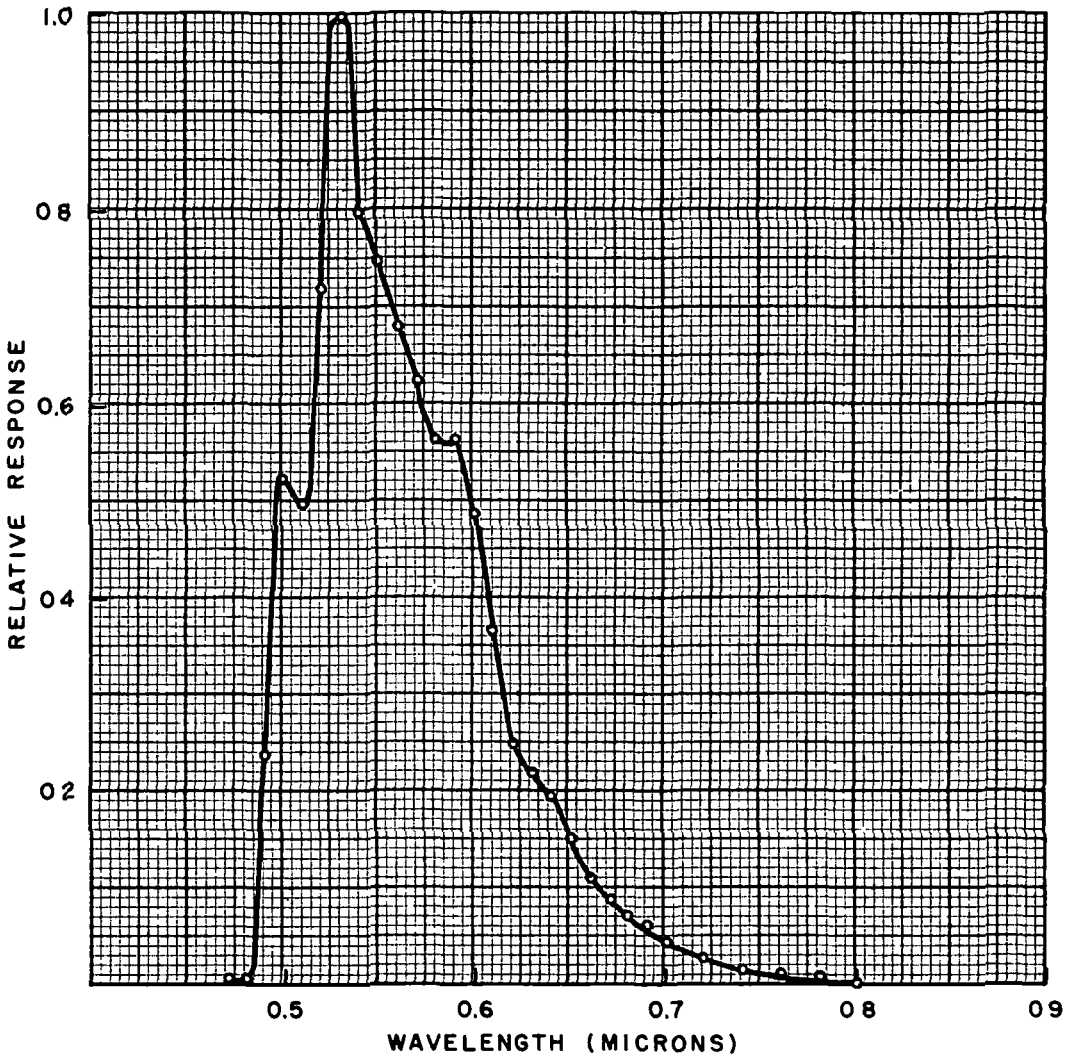


Figure II-12. Camera No. 2, Relative Spectral Response

c. Tape Recorder No. 1

(1) R1 Tape Transport No. 1

ESSA 1 was first equipped with the Serial No. 3-321-06 tape transport as the side I unit. This unit was replaced, however, on December 14, 1964, after failing during thermal-vacuum testing of the integrated spacecraft. The problem was evidenced by the absence of video when side I was commanded in the playback mode. When the tape transport was removed from the spacecraft and examined, it was found that the tape had run off the reel in the record direction. The unit was replaced with the flight-qualified Serial No. 3-321-12 tape transport, and testing was resumed.

The Serial No. 3-321-12 tape transport had successfully passed vibration testing on March 7, 1964, a leak-rate test on March 13, and thermal-vacuum testing on March 16.

During tests of the tape transports on August 27, 1964, this unit was rejected due to poor picture quality at the end-of-tape areas. Flutter and speed variation tests were carried out, but the unit evidenced no departure from the design specifications.

On October 10, 1964, the unit was rejected due to wear of the aluminized end-of-tape leader. At this time the magnetic tape was changed and a 1-foot silvered end-of-tape leader was fitted; in addition, the microswitches and teflon bumpers were reset and the tape heads were cleaned.

As noted, the Serial No. 3-321-12 unit was installed on the spacecraft on December 14, 1964.

On September 2, 1965, system tests of spacecraft OT-3 were interrupted because of an apparent malfunction of the side I tape recorder. The proper command for a remote sequence had been given, and the transport motor had been noted to be making an audible sound in the record mode; however, the system failed to playback upon subsequent command. It was further noted that the power control unit properly switched playback command power to tape recorder No. 2, which performed normally.

Continuity tests established that all relay states and end-of-tape circuits were consistent with an end-of-tape playback condition. When the tape transport was opened for examination, the tape was observed to be at the end-of-tape position on the playback reel with the end-of-tape microswitch fully depressed and the teflon bumpers firmly pressed against the tape on the reel.

The playback reel was mechanically rotated approximately 30 degrees to relieve the pressure on the bumpers and the recorder was then commanded in the remote mode via the normal spacecraft controls. The tape transport functioned normally and was programmed for several record and playback sequences without difficulty.

The settings of microswitches and teflon bumpers were measured; and, although their operating sequence was correct, the settings were very close and occasionally coincided. Typical measured settings relative to the start of the end-of-tape leader were 4-1/2 and 5-1/2 turns of the reel for microswitch and bumper operation, respectively.

With the upper reel full at the end-of-tape position in the record mode, the full reel had to be forced past the teflon bumper to operate the microswitch. Tape build-up was noticeably eccentric, causing cyclic variation in friction torque arising from the microswitch operating leaf spring. The teflon bumper first engaged the tape only 3-1/2 turns of the reel from the start of the end-of-tape leader.

Tape eccentricity in the region of teflon-bumper engagement was measured by dial gauge for both end-of-tape record and end-of-tape playback conditions, clockwise and counterclockwise, and before and after a full record and playback cycle. The peak-to-peak eccentricity was .0045 inch. However, overlaying the end-of-tape leader, in the distance occurring between the operation of the microswitch lever and the engagement of the teflon bumper, a change in eccentricity of .002 inch occurred. After a record and playback cycle, peaks of eccentricity had shifted up to 30 degrees around the reel, and up to .001 inch in amplitude.

It was noted that adhesive had exuded from the boundary between the silvered end-of-tape leader and the associated balancing strip of splicing tape. This had removed a track of oxide from the overlaying turn of magnetic tape.

It was concluded that the microswitches were not effectively backing up the end-of-tape leader sensors. The malfunction could be explained by assuming intermittent operation of the end-of-tape sensing circuit. It was considered possible that the exuded adhesive and attached iron oxide might have caused intermittent contact between sensor and sensing strip, causing an overrun to a jammed condition on the teflon bumper.

At this time all recorders associated with the ESSA 1 test program were examined and reworked as follows:

- The end-of-tape leader length was increased from 1 foot to 4 feet.
- The balancing-tape width was reduced from 1/4 inch to 1/8 inch to eliminate a common boundary with the sensing tape.
- The transport was adjusted to provide 6 ± 2 turns of tape between the start of the end-of-tape leader and microswitch operation and 5 ± 1 turns of tape between microswitch operation and teflon-bumper engagement.

It was also decided to replace the tape, drive belt, and microswitches on tape transport No. 1, after which the unit was requalified.

On November 2, 1965, the microswitches were replaced, on November 4 the tape was replaced with 830 feet of magnetic tape having 4-foot silvered leaders and 1/8-inch wide balancing tape; the microswitches and teflon bumpers were reset in accordance with the new procedure; and the drive belt was replaced and the tension set.

On November 18, after the unit had been cleaned, sealed, and pressurized, leak-rate tests and vibration tests were successfully performed. Thermal-vacuum testing on tape transport No. 1, Serial No. 3-321-12, was completed on November 21,

with satisfactory results, and the unit was reinstalled on spacecraft OT-3 on November 23, 1965.

(2) R2 Power Control Unit No. 1

Spacecraft OT-3 was first fitted with R2 power control unit, Serial No. 3-322-12, on side I. This unit was removed, however, on December 14, 1964, when the side 1 tape recorder failed during thermal-vacuum testing. (As noted earlier, there was no response to a playback command on this occasion; and, upon examination, it was found that the tape had run off the reel in the record direction.)

The Serial No. 3-322-13 power control unit was then installed on the spacecraft as a replacement. This unit had successfully passed vibration testing on April 7, 1964, and thermal-vacuum testing on April 16, 1964.

On September 2, 1965, during a Go-No/Go test of spacecraft OT-3, tape recorder No. 1 failed to respond to a playback command. The tape transport and power control unit were removed from the spacecraft, and the tape was found to be against the bumper at the end-of-tape in the record direction.

Early in the investigation of the malfunction it was concluded that an intermittent failure in the end-of-tape sensing circuit had been the cause of the condition. Since such a failure could have been located in the R2 unit, efforts were made to stimulate the recurrence of the failure in this unit.

The end-of-tape and motor-drive circuits were rapidly cycled through the record/playback cycle. The R2 unit was tested under normal laboratory conditions for 15 minutes at 12 record/playback cycles per second (10,800 cycles in all) without flutter. During this test the unit was repeatedly rotated through 360 degrees in all axes and tapped by hand, hard, on all surfaces with no ill effects.

The test rig was then set up in the environmental test area, and the R2 unit was subjected to low-level random vibration at 0.5 g rms in the 20- to 2000-cps frequency band. The unit was tested for 5 minutes in each of three axes, with no ill effects.

The characteristics of Q2 and Q3 had been impaired by the test rig before the rig had been properly adjusted; but the impairment was not sufficient to affect circuit operation. However, detailed examination of transistor Q2 showed damage which would not be explained by a maladjustment of the test rig. Transistor Q2 energizes the coil of latching relay K5 at end-of-tape playback, and it was considered that an intermittent failure of this circuit would explain the malfunction in tape recorder No. 1.

The design and construction of relay K5 is such that a possible neutral condition of the relay armature can be induced by a combination of electrical conditions within the ratings of the relay. Depending upon the energy content of the coil drive pulse and the pulse duration, contact transfer by the relay armature can be initiated but not completed.

A survey of the Q2 circuit showed that, unlike the case with the Q3 (end-of-tape record) circuits, there is no de-energize surge path for the K5 coil. It was postulated that this resulted in "channeling" of the Q2 chip, causing random large increases in leakage current. Such a condition might produce power pulses to the relay which would be insufficient to cause contact transfer.

As a result of this investigation it was decided to modify the ESSA 1 R2 units to add a protective diode CR-6 across the end-of-tape playback coil of relay K5 and to replace transistor Q2. In addition, each R2 unit would be subjected to about 10,000 operating cycles on the test rig and to partial requalification.

The Serial No. 3-322-11 unit was selected for installation on the spacecraft as R2 power control unit No. 1. This unit had previously passed vibration testing on March 6, 1964, and thermal-vacuum testing on March 19, in each case with satisfactory results. The modifications specified above were performed on November 12, 1965; and the operating cycling test was performed on the same day. Three-axis vibration testing was successfully performed on November 18; thermal-vacuum testing of this unit, in conjunction with tape transport No. 1, Serial No. 3-321-12, was successfully completed on November 21. The R2 power control unit, Serial No. 3-322-11, was installed on spacecraft OT-3 on November 23, 1965.

(3) R3 Signal Electronics Unit No. 1

The R3 signal electronics unit No. 1, Serial No. 3-323-10, successfully passed vibration testing on January 7, 1964, and thermal-vacuum testing on January 8.

d. Tape Recorder No. 2

(1) R1 Tape Transport No. 2

The R1 tape transport No. 2, Serial No. 3-321-13, successfully passed vibration testing on April 8, 1964, a leak-rate test on April 13, and thermal-vacuum testing on April 18.

Tests of the Serial No. 3-321-13 tape transport on August 27, 1964, indicated poor video quality at the end-of-tape areas. Further tests on August 28 showed

the flutter of the unit to be satisfactory; the speed variation, however, was excessive for the last 10 seconds of playback. The unit was opened and adjusted, then cleaned and resealed.

On November 9, 1965, because of the malfunction of the side 1 tape recorder during qualification testing of the integrated spacecraft in September, tape transport No. 2 was removed from the spacecraft and opened for examination. This examination revealed that the aluminized end-of-tape leaders were badly worn and that the end-of-tape sensing bars were being grounded only in an intermittent fashion. Furthermore, at the end-of-tape position in the playback direction, the teflon tape bumper engaged the tape before microswitch operation.

On November 10, 1965, tape transport No. 2 was equipped with a magnetic tape 830 feet in length, 4-foot silvered leaders, and a 1/8-inch wide balancing tape; in addition, the microswitches and teflon bumpers were reset (in accordance with the new procedure referenced earlier in this subsection). On November 19, after being cleaned, sealed, and pressurized, the unit was successfully vibrated in the thrust axis. Leak-rate tests preceded and followed the vibration testing. On November 23, the unit was subjected to temperature tests in a Teeny chamber, with satisfactory results, after which it was reinstalled on the spacecraft.

(2) R2 Power Control Unit No. 2

The R2 power control unit No. 2, Serial No. 3-322-08, initially passed vibration testing on January 7, 1964. Further vibration testing (in the trhust axis only) was necessary, however, in the following month, after the unit had been modified. These modifications consisted of the addition of a 68-ohm resistor and the replacement of a power transistor. The additional vibration test were successfully performed on February 27, 1964; and thermal-vacuum testing of the unit was completed on March 9, with satisfactory results.

On November 12, 1965, as a result of the condition experienced with the side 1 unit, power control unit No. 2 was fitted with the protective diode CR-6 and transistor Q2 was replaced. On November 17, the unit was again subjected to vibration testing, in the thrust-axis only, with satisfactory results. On November 23, 1965, temperature testing of power control unit No. 2 in the Teeny chamber was successfully performed, in conjunction with testing of tape transport No. 2. Both units were reinstalled on the spacecraft on the same date.

(3) R3 Signal Electronics Unit No. 2

The R3 signal electronics unit No. 2, Serial No. 3-323-11, successfully passed vibration testing on January 7, 1964, and thermal-vacuum testing on March 17, 1964.

e. TV Transmitters

The TV transmitters utilized on ESSA 1 were purchased by AED from a commercial manufacturer, and as such, were environmentally qualified before delivery to AED.

When first assembled, the spacecraft included transmitter Serial No. 3-421-04 on side I, and transmitter Serial No. 3-421-05 on side II.

On December 17, 1965, during thermal-testing of the integrated spacecraft at the 0°C level, spurious video modulation from TV transmitter No. 2 produced a herringbone pattern on the TV raster. The spurious frequencies represented a blocking oscillation effect associated with tuning of the crystal oscillator-mixer stage, which produced a distorted modulation output at frequencies in the vicinity of 20 kc. The transmitter was replaced with the Serial No. 3-421-08 unit, which had been tested and had proven to be free of spurious frequencies.

This condition had been first noted during thermal-vacuum testing of TIROS IX. At that time all transmitters had been returned to the manufacturer for examination. An investigation showed that the effect could be found in all units. However, tests of transmitter operation over the range from -10°C to +55°C showed a stable output to be produced when the crystal oscillator was tuned slightly off a point producing spurious frequencies. Based on these results, all units were retuned, and a tuning procedure was established in which the capacitance associated with each crystal oscillator was adjusted to provide an output half-way between the spurious points.

f. Video Coupler

The video coupler, Serial No. 3-427,02, was successfully subjected to temperature tests in the Teeny chamber on March 30, 1964. Environmental testing of the video coupler was achieved with testing of the integrated spacecraft.

g. Computing Triggers

Computing triggers Nos. 1 and 2, Serial Nos. 3-629-04 and 05, respectively, were successfully subjected to vibration testing on May 28, 1964. Thermal-vacuum testing of both units was successfully completed on June 3, 1964.

5. Special Tests

a. Life Expectancy of Tape Transport Wiring Insulation

In January 1966, AED was requested to study the application and expected life of the wiring insulation in the TIROS Series III tape transports used in ESSA 1. From this study it was concluded that the insulation was completely satisfactory for the ESSA 1 mission. The "worst-case" condition was analyzed and determined to provide a minimum continuous operational life of 2400 hours; whereas 1 year of fully operational programming, 6 months on the ground and 6 months in space, would consume approximately 1/5 of this predicted life.

b. Tape-Transport Leak Rate Measurement

In January 1966, AED was also requested to study the leak rate of the TIROS Series III tape transports. A spare unit was utilized and the leak rate was measured at three temperatures (covering the extremes of the qualification temperature profile). The sealing clamp on the transport was then loosened and the seal broken. The unit was repressurized with the nitrogen and helium mixture, and the leak rate was again measured at each of the three temperatures. A life estimate based on these rates, assuming a constant seal characteristic, would indicate a period of approximately 1000 days before the internal pressure would approach the corona levels.

C. PROGRAMMING AND CONTROL SUBSYSTEM

The material relating to this subsystem in presented in Volume II of this report, the classified supplement.

D. TELEMETRY AND TRACKING SUBSYSTEM

1. General

The telemetry and tracking subsystem in ESSA 1 includes two beacon transmitters, two 90-point solid-state telemetry commutators, a signal conditioner, and four subcarrier oscillators.

The ESSA 1 beacon transmitters have a power output of 50 mw; beacon transmitter No. 1 transmits at 136.92 Mc, No. 2 at 136.23 Mc. The beacon transmitters provide two continuous beacon signals to aid in the acquisition and tracking of the satellite by stations of the TIROS ground complex. These signals are also used for the transmission of data to the CDA stations, and the data is applied to the beacon transmitters

through frequency-modulated subcarrier oscillators (SCO's), which amplitude-modulate the beacon carriers. The use of amplitude-modulation on the beacon carriers ensures the presence of the rf carriers for tracking operations, and the absence of sideband components within 1 kc of the carrier. The SCO's associated with each beacon transmitter are as follows:

- Beacon Transmitter No. 1

  No. 1 SCO, 1300 cps (hereafter referred to as SCO 1-1)
  No. 2 SCO, 2300 cps (SCO 2-1)

- Beacon Transmitter No. 2

  No. 1 SCO, 2300 cps (SCO 1-2)
  No. 2 SCO, 1300 cps (SCO 2-2)

Each beacon transmitter receives a continuous flow of attitude data from one of the sensors in the V-head attitude sensor. Normally, the primary SCO's, SCO's 1-1 and 1-2, are constantly powered, and modulated by the V-head attitude-sensor data. Power is applied to the secondary SCO's, SCO's 2-1 and 2-2, only when the satellite receives a command for a direct, playback, or emergency telemetry sequence, and is applied only for the duration of the sequence. In the case of a direct or playback sequence, the data applied as modulation to the secondary SCO's includes back-up attitude data, solar-aspect data, "housekeeping" telemetry data, command-confirmation data, camera-triggering data, and MBC switch stepping data. When the satellite receives an emergency telemetry command, only "housekeeping" telemetry data is applied to SCO's 2-1 and 2-2.

The primary SCO's (SCO's 1-1 and 1-2) were assigned different IRIG channels (channels 5 and 7, respectively) in order to minimize intermodulation between beacons when only attitude data is being transmitted. The secondary SCO on each beacon uses the opposite IRIG channel in respect to the primary SCO.

A control circuit is included in the telemetry and tracking subsystem to permit temporary beacon turn off, upon ground command, for test purposes or for power conservation.

The telemetry and tracking subsystem is shown in block diagram form in Figure II-13. Figure II-14 shows the details of the switching of data to the SCO's and the application of power to the SCO's and the beacon transmitters.

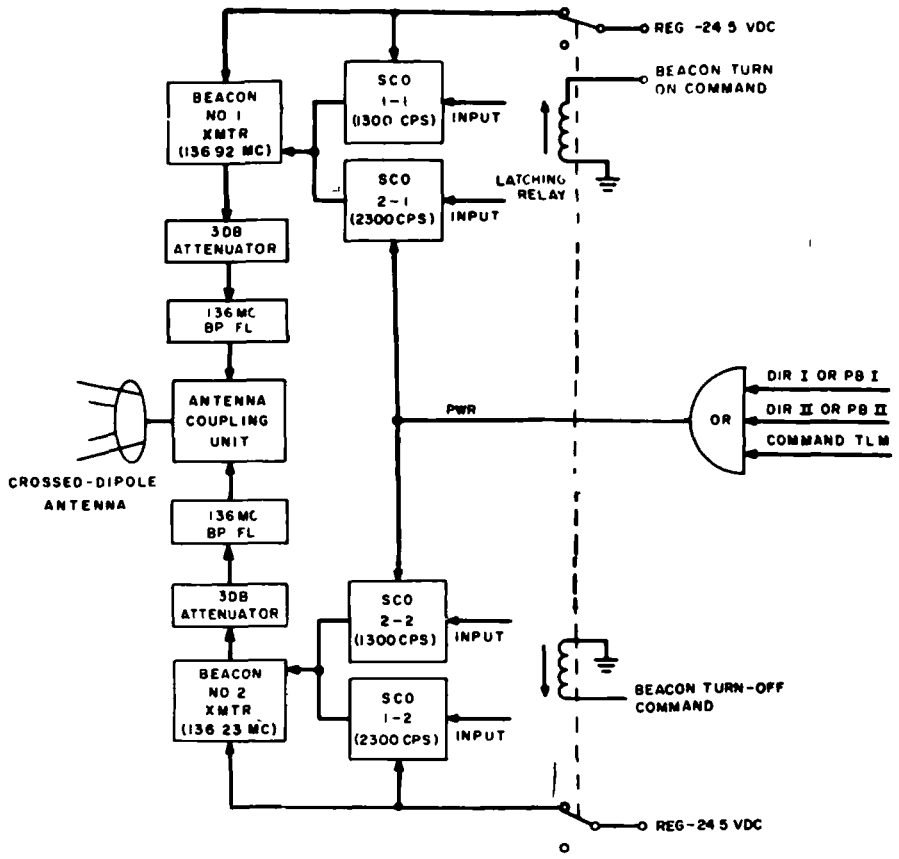


Figure II-13. Telemetry and Tracking Subsystem, Block Diagram

During launch and prior to the separation of the satellite from the third stage of the launch vehicle, the lift-off switch is closed, causing the application of -5.0 volts to the modulation input of SCO 1-1. This, in turn, causes the center frequency of the SCO to be at the 1400-cps level. Upon separation of the satellite from the third stage, the switch opens and the center frequency of the SCO returns to its normal value of 1300 cps, which is an indication to the ground stations that separation has taken place. The indication that the switch has opened is used by the satellite as a timing reference for the automatic activation of the precession dampers and the despin mechanism.

The attitude data applied to the primary SCO's is recorded on a two-channel, Sanborn paper-chart recorder at the CDA stations. The data applied to the secondary SCO's is recorded on a second, identical recorder. This data is transmitted from the satellite in a sequence determined by the command program, as shown in Table II-1. The back-up attitude data applied to each of the secondary SCO's is the same data applied to the respective primary SCO with each beacon.

The two telemetry commutators are automatically energized immediately after warm-up of either camera system and also upon the receipt of an emergency telemetry command

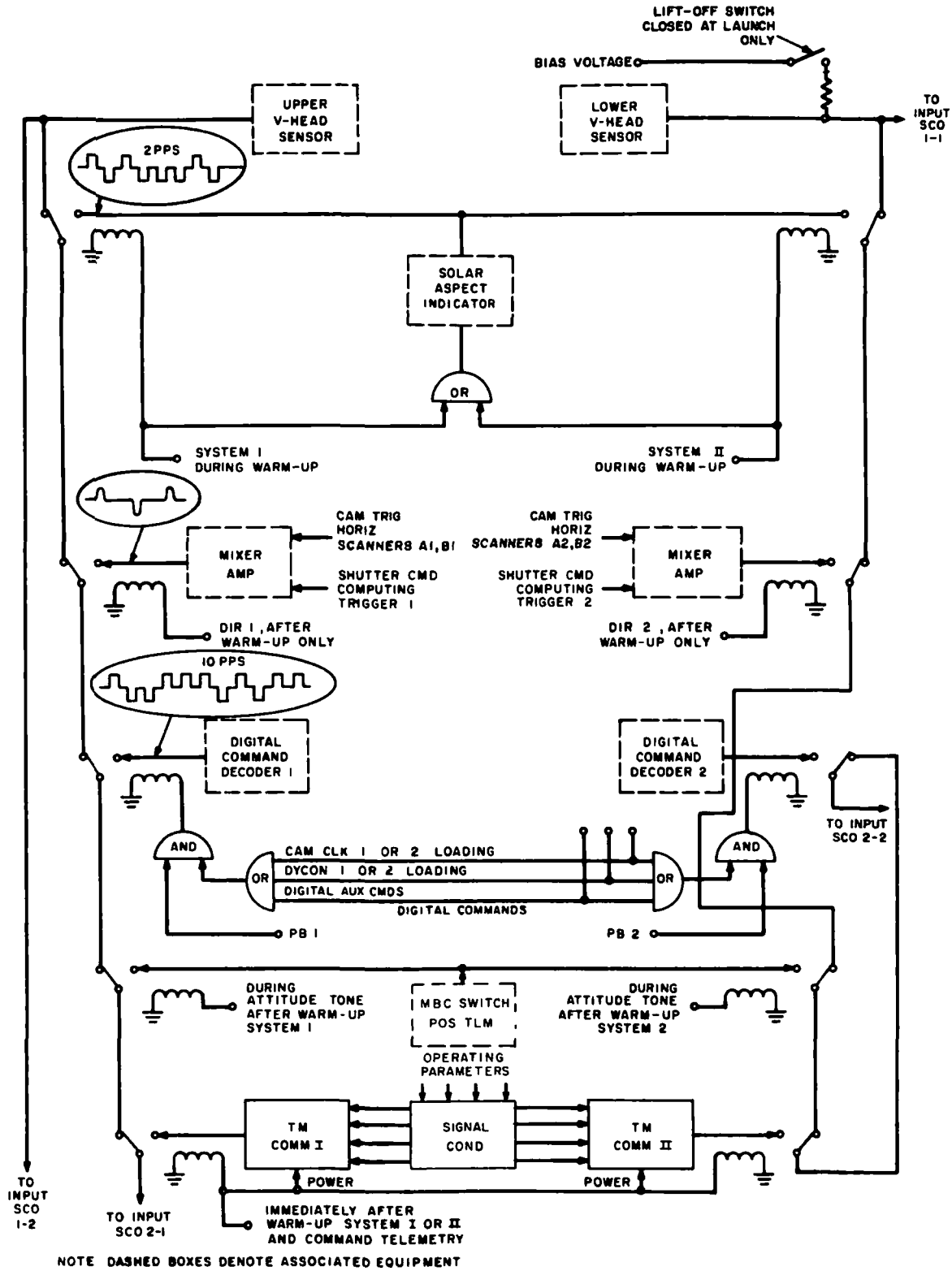


Figure II-14. Telemetry and Tracking Subsystem Data-Switching Arrangement

TABLE II-1. TELEMETRY INPUTS TO SUBCARRIER OSCILLATORS 2-1 AND 2-2

Time Interval	SCO 2-1	SCO 2-2
Before Ground Command NOTE SCO not powered	Upper V-Head Attitude Data	Lower V-Head Attitude Data
During Warm-up of Camera system 1	Solar-Aspect Data	Lower V-Head Attitude Data
Immediately Following Warm-up of First Camera System, 10-second duration	"Housekeeping" Telemetry Commutator No. I	"Housekeeping Telemetry Commutator No. II
During Direct 1, After Warm-up and During "Off" Delay	Camera-Triggering Data from Horizon Scanner Operating with Camera No 1	Lower V-Head Attitude Data
During Playback Sequence, After Warm-up	Upper V-Head Attitude Data	Lower V-HeadAttitude Data
During Playback 1, When FSK Tone-Commands are sent	Command-Confirmation Data (Digital Decoder 1 Output)	Lower V-Head Attitude Data
After Warm-up of either Camera System 1 or 2, When Attitude Tone is Sent	MBC Switch-Position (Subsystem 1)	MBC Switch-Position (Subsystem 2)
During Warm-up of Camera System 2	Upper V-Head Attitude Data	Solar-Aspect Data
During Direct 2, After Warm-up and During "Off" Delay	Upper V-Head Attitude Data	Camera-Trigger Data from Horizon Scanner Operating with Camera No. 2
During Playback 2, After Warm-up	Upper V-Head Attitude Data	Lower V-Head Attitude Data
During Playback 2 When FSK Tone- Commands are Sent	Upper V-Head Attitude Data*	Command-Confirmation Data (Digital Decoder 2 Output)
Whenever Emergency Telemetry Command is Programmed	"Housekeeping" Telemetry, Commutator No. I	"Housekeeping" Telemetry, Commutator No. II.

*This data is recorded on a magnetic tape recorder at the CDA station, but not on the Sanborn recorder, where it is replaced by a recording of the command as transmitted to the satellite.

from a CDA station. Each commutator samples 88 channels at a 10-pps rate, after a warm-up period of approximately 2 seconds. During the 2-second warm-up period, the output of each commutator (nominally, minus 5 volts) is applied to the primary SCO's. The trailing edge of this signal (the end of the 2-second warm-up period) is the start of channel 1 of the "housekeeping" telemetry. When the warm-up periods of the two commutators differ, initiation of telemetry readout by the two commutators is not simultaneous. As noted, the normal operating rate of each commutator is 10 pps; however, the unit would run at a rate of 9 pps in the event that the external sync pulse should be lost.

The first four channels on each commutator are calibration voltages which are supplied to facilitate interpretation of the telemetered data. The channel assignments for each commutator cover the same sampling points; however, except for channels 1 and 2, the first two calibration channels, the sampling points are covered in two different sequences, with the consequent advantages of redundancy and reliability in commutator operation. Table II-2 lists the channel assignments for each commutator.

2. Equipment Description

a. Beacon Transmitters

The ESSA 1 beacon transmitters are identical to the units used on previous TIROS satellites. Beacon transmitter No. 1 has a carrier frequency of 136.92 Mc and beacon transmitter No. 2, 136.23 Mc. The transmitters are crystal-controlled, with a carrier frequency stability of 0.005 percent.

The power input to the beacon transmitters is supplied at -24.5 volts dc, and the output power for each unit is 50 milliwatts with the transmitter modulated by at least one SCO. The beacon transmitters are designed to operate with a 50-ohm load, and the output signal from each unit is an amplitude-modulated RF signal. The depth of modulation on either transmitter is more than 40 percent when the transmitter is modulated by one SCO and less than 100 percent when modulated by two SCO's.

b. Subcarrier Oscillators

The SCO's used in the telemetry system are voltage-controlled oscillators which operate with an input power level at -24.5 volts, and a total power drain for each unit of 20 milliamperes.

The input signal level of the telemetry data supplied to the SCO's is 0 to -5 volts, with (1) 0 volts producing the lowest frequency, (2) -2.5 volts, the center frequency, and (3) -5.0 volts the highest frequency. The output signal is a-c coupled and adjustable from 0 to 1.5 volts peak-to-peak into an 8000-ohm load.

TABLE II-2. CHANNEL ASSIGNMENTS FOR ESSA 1 TELEMETRY COMMUTATORS

Channel No.		Assignment
Comm. I	Comm. II	
1	1	Calibration 0 volts
2	2	Calibration -5 volts
3	4	Calibration -3.5 volts
4	3	Calibration -1.5 volts
5	61	"X" Battery-string voltage
6	62	"Y" Battery-string voltage
7	63	"Z" Battery-string voltage
8	64	Load-bus voltage
9	65	MASC coil
10	10	CHANNEL 10 MARKER
11	57	Solar-cell array voltage
12	58	Solar-cell patch voltage
13	59	Solar-cell array current
14	66	-24.5 volt reg No. 1
15	67	-24.5 volt reg. No. 2
16	68	-13.0 volt reg. No. 1
17	69	-13.0 volt reg. No. 2
18	51	Side No. 1 warmed-up
19	52	Side No. 2 warmed-up
20	20	CHANNEL 20 MARKER
21	53	Side No. 1 direct sequence
22	54	Side No. 2 direct sequence
23	55	Side No. 1 playback sequence
24	56	Side No. 2 playback sequence
25	81	TV transmitter, No. 1 or 2
26	82	Recorders isolated or crossed
27	83	Horizon scanner pair A or B
28	84	DYCON No 1, mode indicator A
29	85	MASC coil
30	30	CHANNEL 30 MARKER
31	71	Time base generator No. 1
32	72	Time base generator No. 2
33	73	Camera clock No 1 power
34	74	Camera clock No. 1 count enable
35	75	Camera clock No 1 special count enable
36	76	Camera clock No. 2 power
37	77	Camera clock No 2 count enable
38	78	Camera clock No 2 special count enable
39	79	Camera clocks isolated or crossed
40	40	CHANNEL 40 MARKER
41	5	Computing trigger No. 1 power
42	6	Computing trigger No. 2 power
43	7	DYCON No 1 power
44	8	DYCON No. 1 command load enable
45	9	DYCON No. 1 count enable

TABLE II-2. CHANNEL ASSIGNMENTS FOR ESSA 1 TELEMETRY COMMUTATORS (Continued)

Channel No.		Assignment
Comm. I	Comm. II	
46	11	DYCON No. 1 mode indicator B
47	12	DYCON No. 1 mode indicator C
48	13	DYCON No. 1 mode indicator D
49	45	MASC coil
50	50	CHANNEL 50 MARKER
51	14	DYCON No. 2 power
52	15	DYCON No. 2 command load enable
53	16	DYCON No. 2 count enable
54	17	DYCON No. 2 mode indicator B
55	18	DYCON No. 2 mode indicator C
56	19	DYCON No. 2 mode indicator D
57	86	QOMAC-coil state
58	87	MBC polarity and QOMAC high-torque power enable to MBC switch
59	88	MBC switch position
60	60	CHANNEL 60 MARKER
61	21	Rocket switch position
62	22	Receiver No. 1 AGC
63	23	Receiver No. 2 AGC
64	24	Vidicon No. 1 high voltage
65	25	Vidicon No. 2 high voltage
66	26	Filament and focus current No. 1
67	27	Filament and focus current No. 2
68	28	500-cps converter No. 1
69	29	500-cps converter No. 1
70	70	CHANNEL 70 MARKER
71	31	Opposite commutator power
72	32	DYCON No. 2 mode indicator A
73	33	High-level charge, ON-OFF
74	34	Temperature, QOMAC Coil*
75	35	Temperature, side hat, 6 in from baseplate**
76	36	Temperature, side hat, 6 in from top**
77	37	Temperature, hat, at 2-5/8 in radius**
78	38	Temperature, hat, at 3 in radius**
79	39	Temperature, hat, at 12 in radius**
80	80	CHANNEL 80 MARKER
81	41	Temperature, battery pack*
82	42	Temperature, camera No. 1*
83	43	Temperature, camera No. 2*
84	44	Temperature, TV trasnmitter*
85	46	Temperature, beacon transmitter*
86	47	Temperature, camera clock No. 2*
87	48	Temperature, voltage regulator No. 2*
88	49	Temperature, baseplate, at (approximately) 10° radial*
89	89	NOT AVAILABLE FOR TELEMETRY
90	90	NOT AVAILABLE FOR TELEMETRY

*Narrow-Range Sensor
**Wide-Range Sensor

Standard IRIG subcarrier frequencies, with deviation limits of ±7.5 percent of center frequency, are employed. The primary SCO for beacon transmitter No. 1 is a Channel 5 (1300 cps) unit; the primary for transmitter No. 2 is a Channel 7 (2300 cps) unit. The secondary SCO's for the beacon transmitters have the opposite IRIG Channel assignments. (The SCO's on each channel are electrically identical units.)

c. Beacon/SCO Assembly

The two beacon transmitters and the four SCO's comprise a single assembly. The beacon transmitters are mounted within the assembly; the SCO's however, are mounted on top of the assembly housing to permit accessibility to the output amplitude adjustment. The circuit board mounted within the assembly housing carries the "OR" gate power diodes for the secondary SCO's and various terminals used as tie points for harnessing internal to the assembly.

d. Telemetry Commutators

The telemetry commutators used on ESSA 1 are 90-channel, solid-state devices identical to those used on TIROS IX. As shown in the block diagram in Figure II-15, the 90-channel commutator consists of (1) five 16-channel modules and one 8-channel module, (2) frame and pedestal insertion circuitry, (3) associated logic and sync circuitry, and (4) a power supply.

Channel selection in the basic modules is achieved by coincidence of negative going x and y levels. When x or y or both levels are positive, the gate is non-conducting. When x and y simultaneously are negative, the diodes are reverse biased. A current drive of approximately 50 microamperes flows from constant current generator No. 1 through the collector and the base to the transistor gate being turned ON to constant current generator No. 2. The net unbalance between currents generators No. 1 and No. 2 represents back-current and is nominally zero. The input impedance of the switch is determined by the output impedance of the two constant current sources and the input impedance of the buffer.

The channel sampling sequence is derived from a binary counter. The outputs of the first four stages, labeled "x" counter, are fed into a diode matrix, and 16 sequential outputs are obtained. The outputs of the last three stages, labeled "y" counter, are similarly fed into another diode matrix and six sequential outputs are obtained.

The x and y counters are driven by an internal clock. The clock can be synchronized by the spacecraft 10-pps waveform.

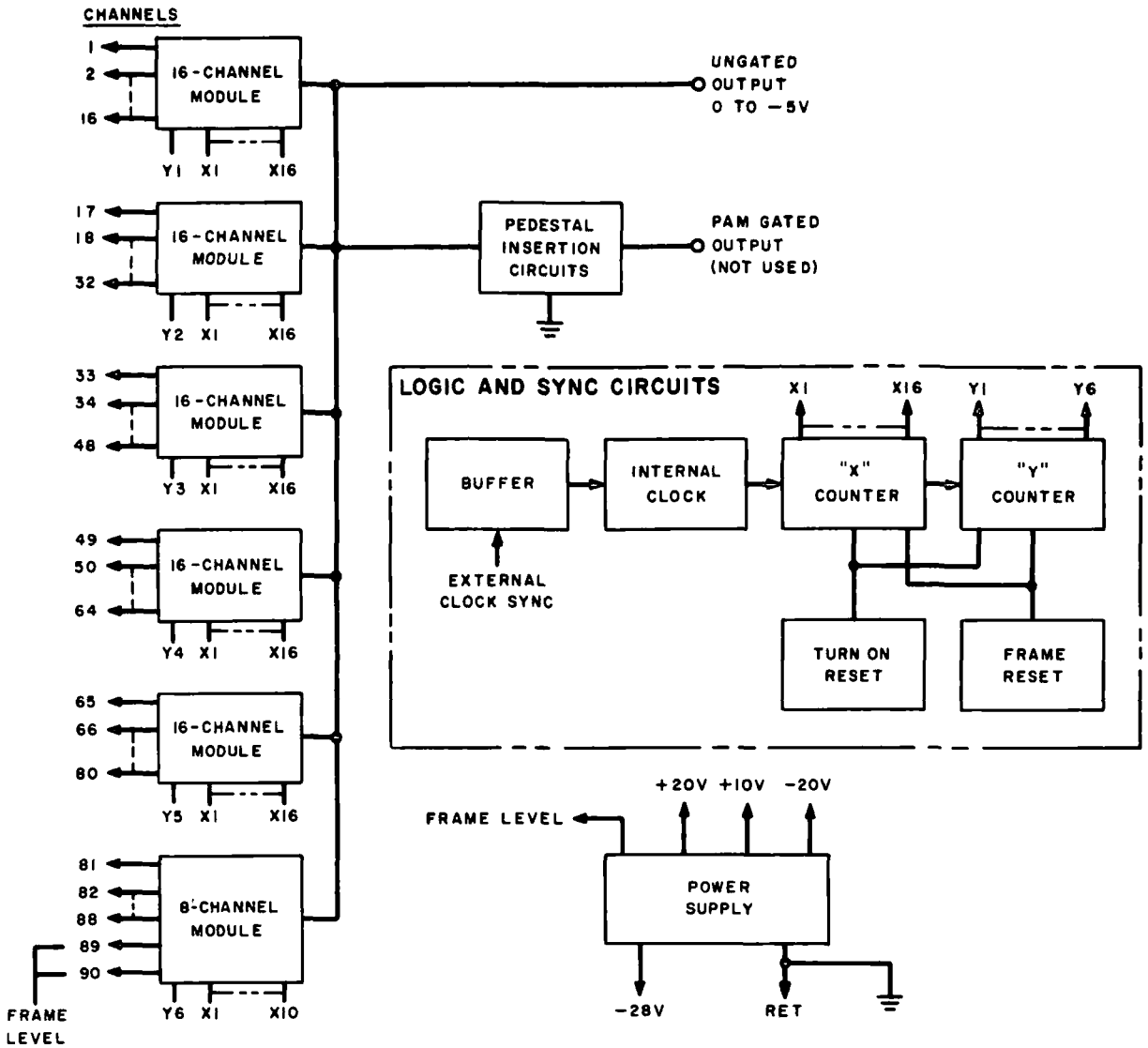


Figure II-15. Telemetry Commutator, Block Diagram

During sampling of channels 89 and 90, a frame pulse is generated in the signal generator for the commutators. This signal drives the relay to remove power from the commutators at the completion of frame 90.

When power is applied, a monostable multivibrator is provided to reset and hold the counters for 2 seconds. At the completion of the 2-second period, the commutator starts at channel one.

The telemetry commutator has two outputs; an ungated output (non-return to zero). The ungated output signal is from 0 to -5 volts. The gain of the buffers is approximately unity (0.9975). The gated output signal, following the insertion of the pedestal circuits,

has an output signal from 0 to -5 volts. During the OFF period, the output is at 0 volts. During the ON period, the output is -1 volt for an input signal of zero volts and -5 volts for an input signal of -5 volts. The gated output was not used on ESSA 1.

The power supply consists of a dc-dc inverter and two regulators. The secondary voltages are each regulated through a stable, high-gain open-loop series regulator.

A component reliability problem* in the telemetry commutators was experienced, in December 1964, during environmental testing of spacecraft OT-3. To eliminate this problem, the commutators were modified as follows:

(1) a short was effected across a wire-wound resistor that appeared prone to mechanical weakness because of the filing technique used to set its resistance;

(2) another short was effected across a diode that had been installed to provide reverse-polarity protection during testing; and

(3) an externally mounted, 10-ohm resistor was placed in series with the power input to the commutator to replace the original wire-wound resistor.

e. Signal Conditioner

The function of the signal conditioner is to buffer telemetry signals from the sampling points and provide output levels to the commutators which are compatible with subsystem requirements, i.e., within the range of 0 to -5 volts. This is acccomplished either by voltage division or amplification. Those telemetry signals which are suitably buffered before being applied to the signal conditioner are directly connected to the proper connecting pins in each commutator.

All input signals to the signal conditioner are supplied in pins on three 37-pin connectors and one 15-pin connector. All outputs from the signal conditioner are applied to each commutator in pins of two 50-pin connectors. (Each telemetry commutator is a 90-channel unit; the additional connections are for the functional inputs for the signal conditioner and the telemetry commutators.)

Unregulated power (-28 volts) is continuously available to the signal conditioner, and separate power lines are provided to supply unregulated power (-28 volts) to each commutator during the period the commutators are being operated. Whenever commutator power is supplied, a -24.5 volt power source is applied to the signal conditioner.

*Detailed under "History of Qualification Testing."

An external 10-pps signal is applied through the signal conditioner to the commutators to provide the sampling rate. In addition, two circuits are provided for shut-off of the commutators. These circuits drive relays external to the signal conditioner. The circuits are continuously powered by means of the unregulated -28 volt power sources, and the relays are connected to this source.

The signal conditioner also provides the -3.5, and -1.5 volt calibration signals for each commutator. The remaining calibration signals (0 volt and -5 volts) are supplied from within the commutators. The signal conditioner also provides for the application of a -5 volt reference signal to channels 10, 20, 30, 40, 50, 60, 70, and 80 of each commutator. In addition, each commutator chassis ground is connected to the chassis of the signal conditioner.

3 Testing

a. Beacon/SCO Assembly

The ESSA 1 beacon/SCO assembly, Serial No. 3-423-03, is comprised of the following units:

(1)	beacon transmitters:	No. 1	Serial No. 3-405-05
		No. 2	Serial No. 3-406-05
(2)	subcarrier oscillators:	1-1	Serial No. 1203
		2-1	Serial No. 1210
		1-2	Serial No. 1211
		2-2	Serial No. 1205
(3)	diode board assembly.		Serial No. 1758203

The beacon transmitters and the SCO's were purchased from commercial manufacturers by AED. The units were, consequently, subjected to qualification testing by the manufacturers and bench testing at AED. The units were then integrated into the assembly, and the assembly was subjected to qualification testing before being integrated on the spacecraft. The beacon/SCO assembly was successfully subjected to vibration testing on July 14, 1964, and thermal-vacuum testing was completed on July 17 with satisfactory results.

The operation of the beacon transmitters was successfully tested in December 1965 during tests of the beacon transmitters and the modified RF coupling system (further details on these tests are presented in Appendix B).

b. Telemetry commutators/Signal Conditioner Package

The telemetry commutators (Serial Nos. 299 and 304) and the signal conditioner (Serial No. 3-925-03) were purchased items which were tested by the manufacturer prior to delivery to AED. At AED, the units were integrated into a single package and, as such, subjected to environmental qualification testing. Vibration testing was successfully performed on June 16, 1964, and thermal-vacuum testing was successfully completed on June 19.

On January 3, 1965, because of failures experienced with the telemetry commutators during environmental qualification testing of the integrated spacecraft, the commutators were modified as noted earlier in the functional description of these units.

After the modifications to the commutators had been effected, they were integrated with the signal conditioner, and the package was subjected to vibration testing on January 4, 1965, and thermal testing in the Teeny chamber on January 7. With the successful completion of these tests, the package was declared qualified for integration with the spacecraft.

E. REFERENCE-INDICATOR SUBSYSTEM

1. General

The reference-indicator subsystem in ESSA 1 is comprised of the V-head attitude sensor, the horizon scanners, and the solar-aspect indicator. These units were purchased from manufacturers outside AED and were qualified prior to the delivery to AED.

The horizon scanners provide an indication of the sky-earth transitions during a rotation of the satellite. As such, this data presents an indication of the satellite spin period, and is used primarily in camera triggering and in programming operation of the MASC to increase or decrease the spin rate. In addition, this data can be used to determine the satellite attitude.* Horizon-scanner data is transmitted to the interrogating CDA ground station during a direct picture-taking sequence.

The V-head attitude sensor is the basic system used to obtain an indication of the satellite attitude. This data is used in programming the operation of the QOMAC coil in both the performance of the initial wheel-orientation maneuver and in the maintenance

*A detailed discussion of the primary and secondary methods of measuring the satellite's attitude is presented in "Attitude Handbook, for the TIROS IX Meteorological Satellite System", Astro-Electronics Division, Radio Corporation of America, M-2036, 7 October 1964.

of the wheel attitude. Data from the V-head attitude sensor is continuously transmitted to the ground.

The solar-aspect indicator provides measurements of the γ angle, i.e., the angle between the satellite-sun line and the satellite spin axis. Knowledge of the γ angle is required for power-availability studies, and, in certain instances, is used in determining satellite attitude. Solar-aspect indicator data is transmitted to the CDA ground stations while a commanded camera system is in the 28-second warm-up period. During this time, three or four measurements of the γ angle are transmitted to the ground station.

2. Horizon Scanners

a. General

ESSA 1 contains four IR horizon scanners, comprising two pairs, A and B. As noted previously, the horizon scanners are located on the baseplate in a specific relationship to the cameras. This relationship is such that when the optical axis of the selected camera is aligned with the local vertical (assuming a perfect Wheel attitude), the horizon scanners for the "side", i.e., one scanner of each pair, are viewing the horizon, in a sky-earth transition.

Horizon scanners A1 and B1 are associated with the camera system No. 1; Figure II-16 is a logic diagram for the operation of these scanners. (Scanners A2 and B2, which are associated with camera system No. 2, operate in identical fashion.)

The command for a picture-taking sequence also includes the selection of one horizon-scanner pair (leaving the second pair available as a redundancy feature). Consequently, if camera system No. 1 and horizon scanner pair A are selected, the horizon-crossing pulse for the camera-triggering units associated with camera No. 1 is supplied by scanner A1. Scanner A2 simultaneously supplies camera system No. 2 with camera-triggering data, though the command did not call for the application of power to camera No. 2.

As shown in Figure II-16, the -24.5 volt and -13 power for scanner A1 are supplied from the command distribution unit. In the case assumed above, the output from the horizon scanner A1 produces several outputs from the associated threshold amplifier, as follows:

(1) an a-c coupled output is supplied to the computing trigger for the selected camera system, and, thence, to the telemetry subsystem.

(2) an a-c coupled output is also supplied to a test point.

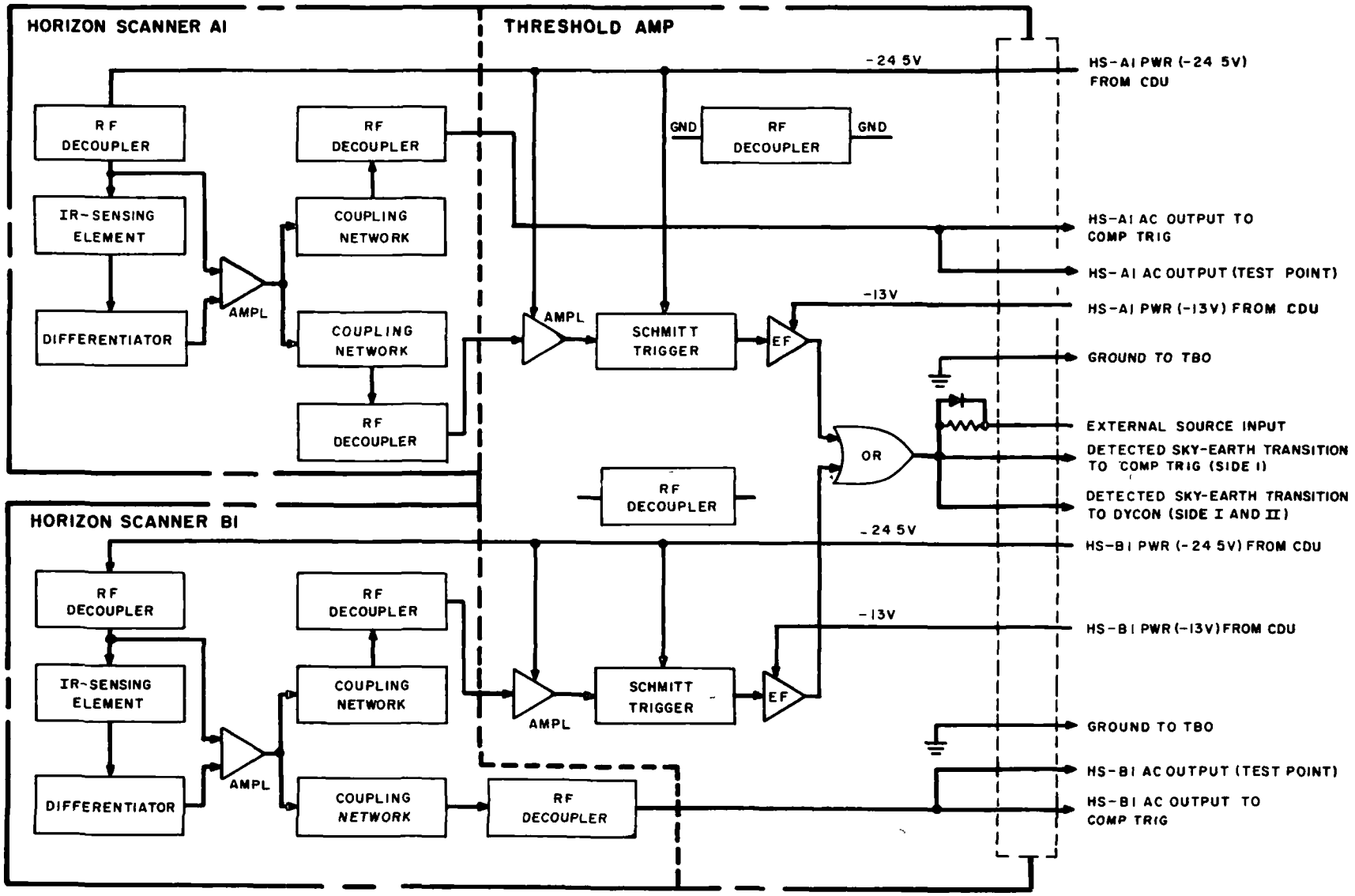


Figure II-16. Logic Diagram of Horizon Scanners A1 and B1

(3) the detected sky-earth transition is supplied to the computing trigger for the selected camera system.

(4) the detected sky-earth transition is supplied to both Dycon units.

b. Equipment Description

Each horizon scanner consists of (1) a thermister bolometer, (2) a low-noise, high-gain amplifier, and (3) a filtered d-c source which provides a bias for the bolometer and amplifier. Each bolometer contains two, matched, emission flakes, sensitive to the 1.8- to 23-micron IR range and connected in series across the bias.

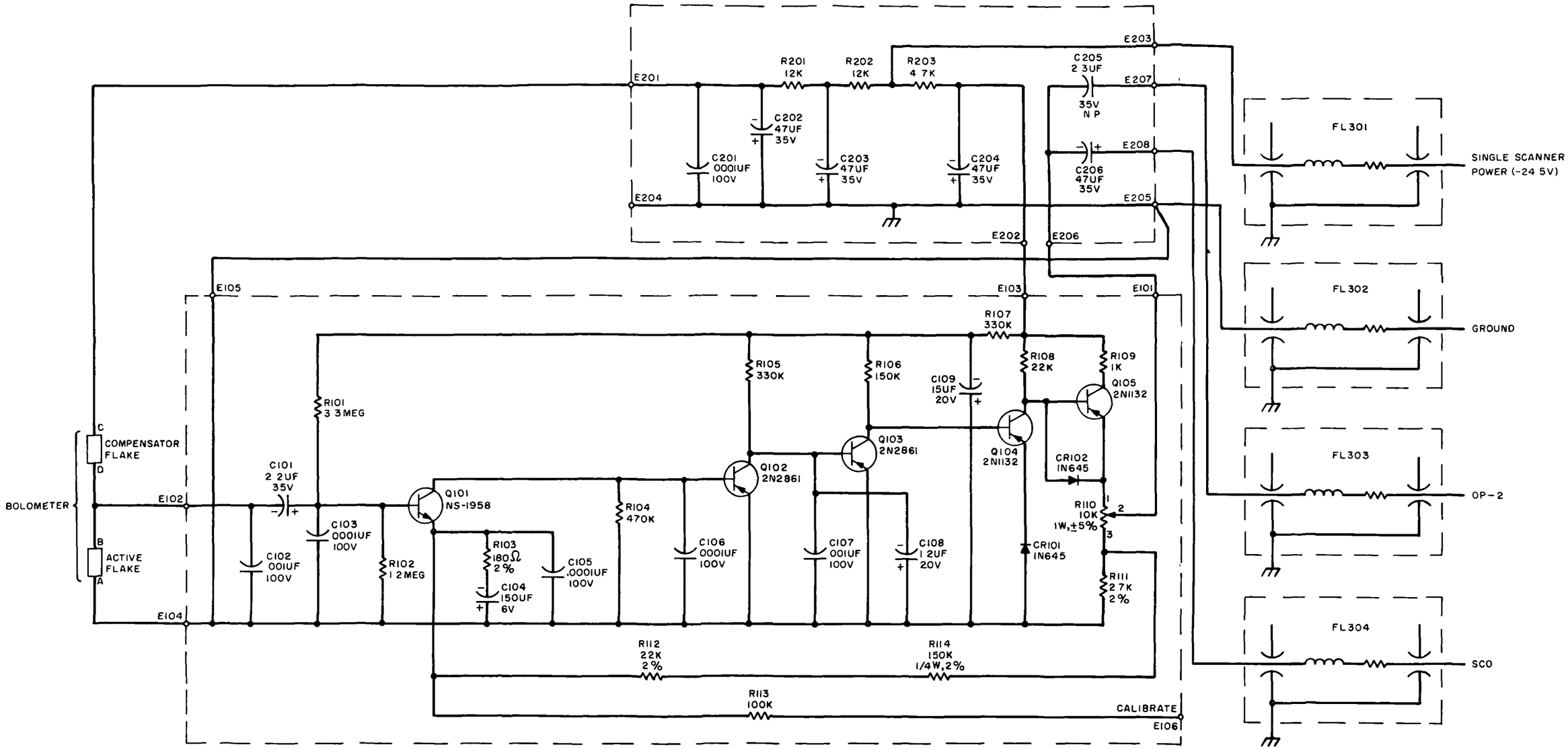
One of the flakes (called a compensator) is shielded from radiation while the other flake (the active flake) is exposed to radiation. When the active flake is heated by radiation from the earth, its resistance changes slightly with respect to the compensator. This produces a small (typically, 300 microvolts peak signal for 220°K equivalent earth temperature) voltage change that is amplified to a useful level of several volts peak.

Rather than acting as an absolute radiometric device, the bolometer indicates when a change in radiance is viewed by the scanner. The imbalance in the d-c bias voltage at the node between the flakes is 100 or more times greater than the signal due to radiance, and there is a change in input as the satellite spins and the scanner "scans" from the sky across the earth horizon. The change in d-c bias voltage is differentiated, with a positive-going transient being produced for sky-earth transitions, and a negative-going transient being produced for earth-sky transitions.

Table II-3 is a summation of the parameters of both the horizon scanners and the V-head attitude sensor.

Figure II-17 is a schematic of the horizon-scanner amplifier. This type of amplifier is designated as a "starved transistor", low-noise amplifier because the first-stage Q 101 transistor is operated at extremely low (1 μ amp) collector currents

The remainder of the amplifier is direct-coupled to minimize RC time constants in the overall system, since such constants could cause difficulty at low frequencies. At the low frequency cut-off required for the sky-earth crossing slopes, (0.8 cycles for the V-head attitude sensor and 3 cycles for the horizon scanners), most of the time constants are eliminated in a direct-coupled configuration to prevent them from coinciding. In the amplifier used in this application, three capacitors, with their associated equivalent circuit resistance, control the low-frequency roll-off of the amplifier. These capacitors are C101, C104, and C206, with C104 and R103 controlling the time constant.



NOTE

ALL RESISTORS ARE 1/4 W, ± 5% UNLESS OTHERWISE SPECIFIED

Figure II-17. Schematic Diagram of Horizon-Scanner Amplifier

TABLE II-3. PARAMETERS OF ESSA 1 HORIZON SCANNERS AND V-HEAD ATTITUDE SENSOR

Bolometer	
Type	Immersed Thermistor
Active Flake Area	0.075 x 0.075 mm (masked)
Peak allowable Bias at 25°C	29 volts/flake
Anti-Reflection Coating	Peaked at 10 microns
Time Constant	2 ms
Responsivity	
Resistance	225 Kohms each flake
Noise	0.6 rms (5 to 100 cycle bandwidth)
Field-of-View	1.3° by 1.3° ± 20% (1/2 power point)
Spectral Response	2.0 to 20 microns
Amplifier	
Type	Low-Frequency, Differentiating
Noise Figure	5 db
Gain (typically)	3000 adjustable ± 30%
Bandwidth (3 db points)	
Horizon Scanner	3.0 to 45 cps
V-head Attitude Sensor	0.8 to 15 cps
Overshoot	5% of peak-pulse amplitude
Recovery to Overload	< 1 sec
Current Drain	3 ma, max
System	
Volts for 220°K Equiv. Earth Temp	0.9 v peak
Amplitude Variation with Temp (-10°C to 40°C)	± 15%

DC stability of the amplifier is achieved by negative feedback of the voltage across R111 to the input stage. Most of this feedback is bypassed by R103 and C104 to provide a-c gain, which is approximately 3000, while d-c gain is approximately 100.

Capacitors C102, C103, C105, C106, C201, together with feed-thru filters FL301 through FL304, provide attenuation of RF interference, particularly from the beacon transmitters.

Capacitor C108 "rolls-off" the low-frequency response of the amplifier, starting at 15 cycles, to limit the noise bandwidth and to control the phase shift for amplifier stability.

3 V-Head Attitude Sensor

a. General

The V-head attitude sensor consists of two IR sensors and associated electronics. The two IR sensors comprise two "heads", arranged in a "V" configuration, as shown in Figure II-18. The plane formed by the optical axes of the two heads contains the satellite spin axis, and the bisector of the angle between the two heads is normal to the spin vector. As the satellite spins, the optical axes of the two heads trace out two conic sections in space. The intersection of these paths with the earth, and the corresponding outputs from the heads, are depicted in Figure II-19, A. When the satellite is in the perfect Wheel attitude, the spin axis is normal to an earth radius, and the outputs of the two heads are identical, as shown in Figure II-19, B. A roll error is detectable as an inequality in the pulse durations from the two heads, as shown in Figure II-19, C.

The orientation of the spin axis is determined from attitude sensor data derived over a discrete section of the orbit. The accuracy of the computation is determined by the resolution and reliability of each of the data points. Consequently, at least 10 minutes of data from a pass over a given CDA station is required for accurate determination of spin-axis attitude using V-head sensor data alone.

b. Equipment Description

The V-head attitude sensor is functionally similar to the horizon scanners. The IR sensors used in the attitude sensor are the same type as those used in the horizon scanners, and the amplifiers for the two systems are also quite similar.

The major difference between the amplifier used for the V-head attitude sensor and that used with each horizon scanner is represented by the active filter used in the

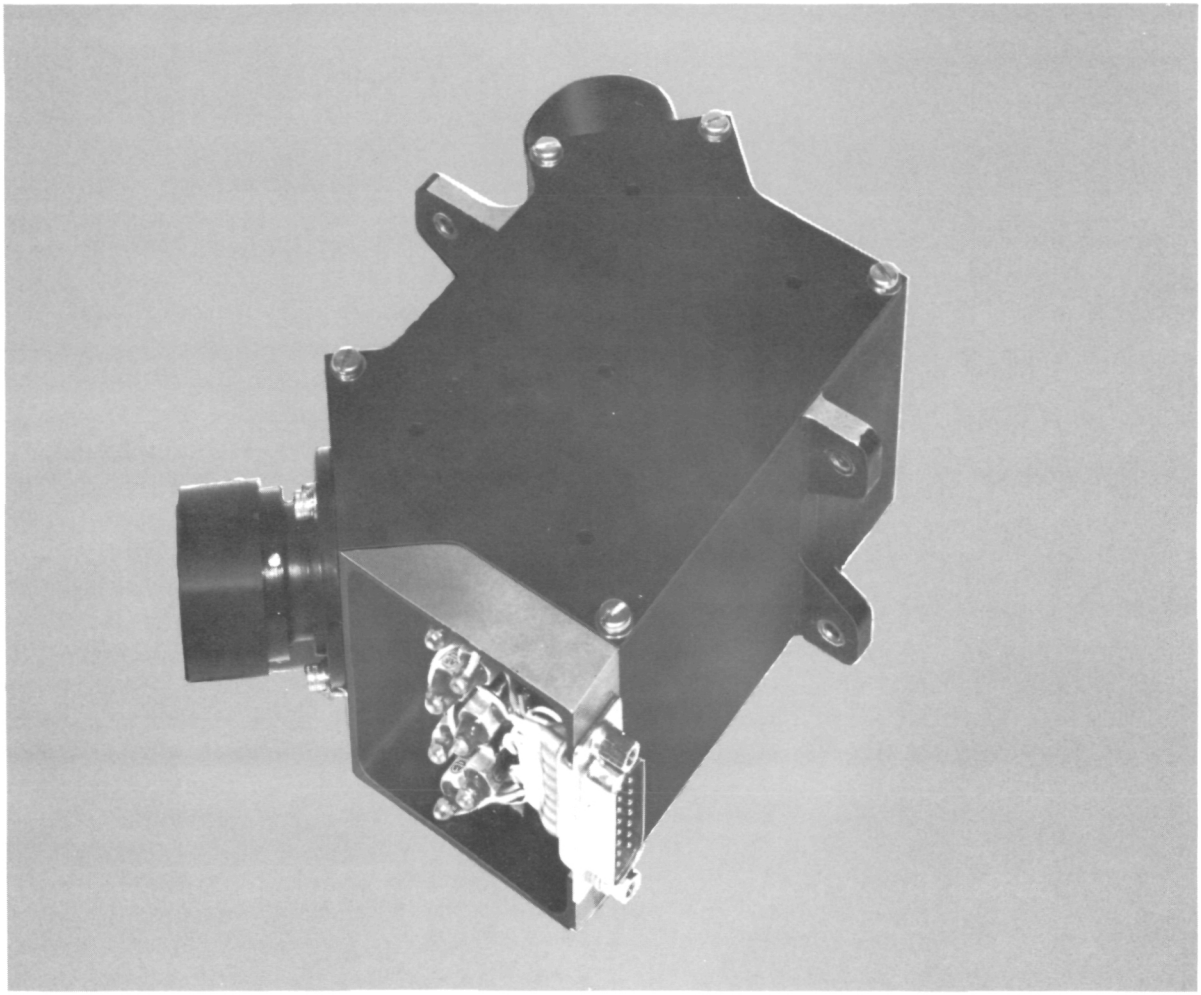


Figure II-18. V-Head Attitude Sensor, with Cover Removed

power circuit of the attitude sensor to move the low-frequency roll-off of the power circuit below that of the amplifier. No filter was required in the horizon-scanner amplifiers, which have a 3-cycle cut-off frequency, as compared to the 0.8-cycle cut-off frequency of the amplifier in the V-head attitude sensor.

A further difference between the V-head attitude sensor and the horizon scanners is the 8-micron "cut-off indium antimonide filter used on the sensing elements of the V-head attitude sensor to reduce sun interference. This reduction is accomplished without an appreciable effect on the earth signal, since at the 200°K earth temperature only a small percentage of the earth energy is below 8 microns. The insertion loss of the filter, itself, accounts for the major portion of any signal loss; and compensation for the insertion loss is obtained in the amplifier, which has a gain figure approximately 50 percent greater than that of a horizon-scanner amplifier. Figure II-20 is a logic diagram of the V-head attitude sensor. The attitude sensor is supplied -24.5 volt d-c power from the auxiliary control unit.

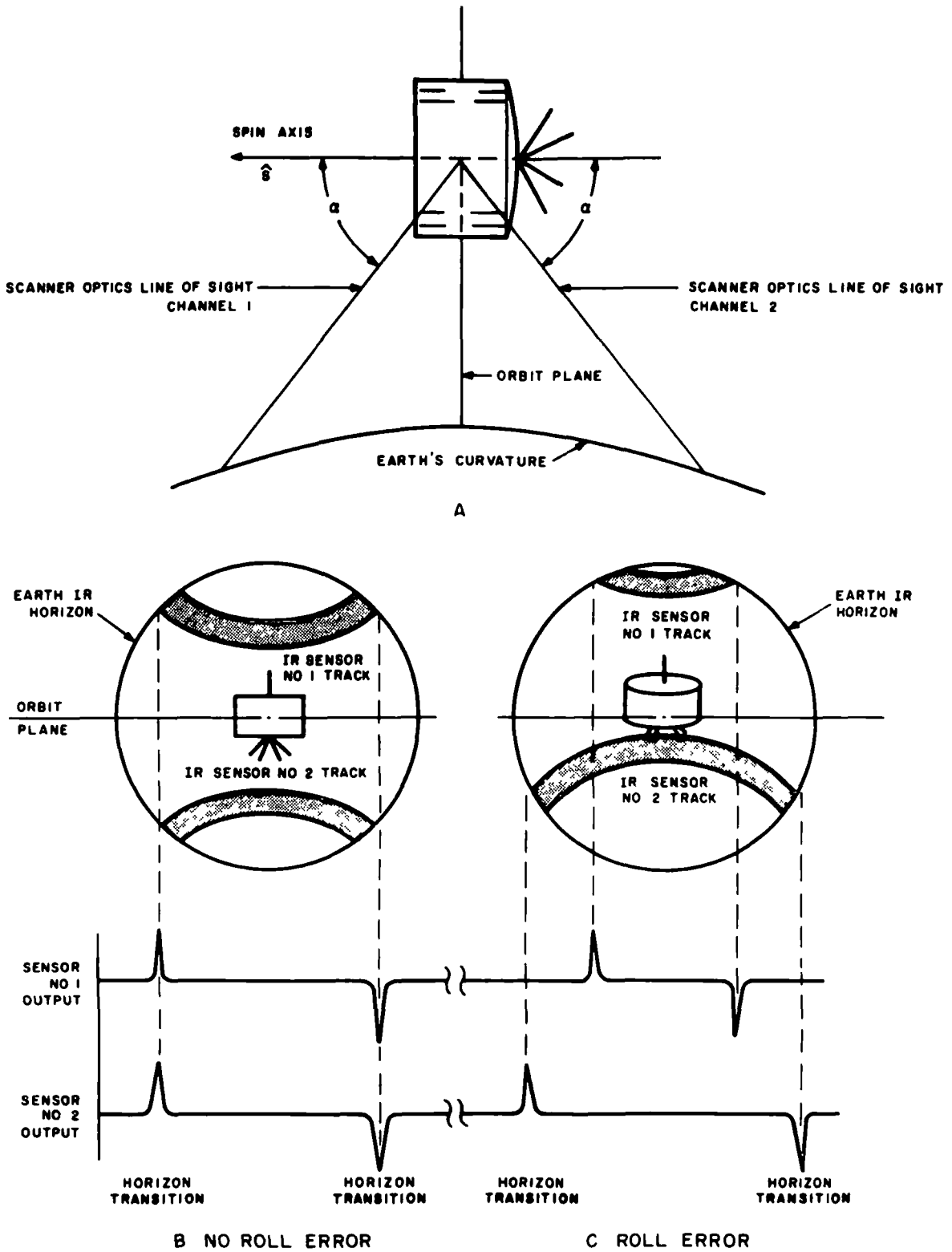


Figure II-19. V-Head Attitude Sensor Geometry and Outputs

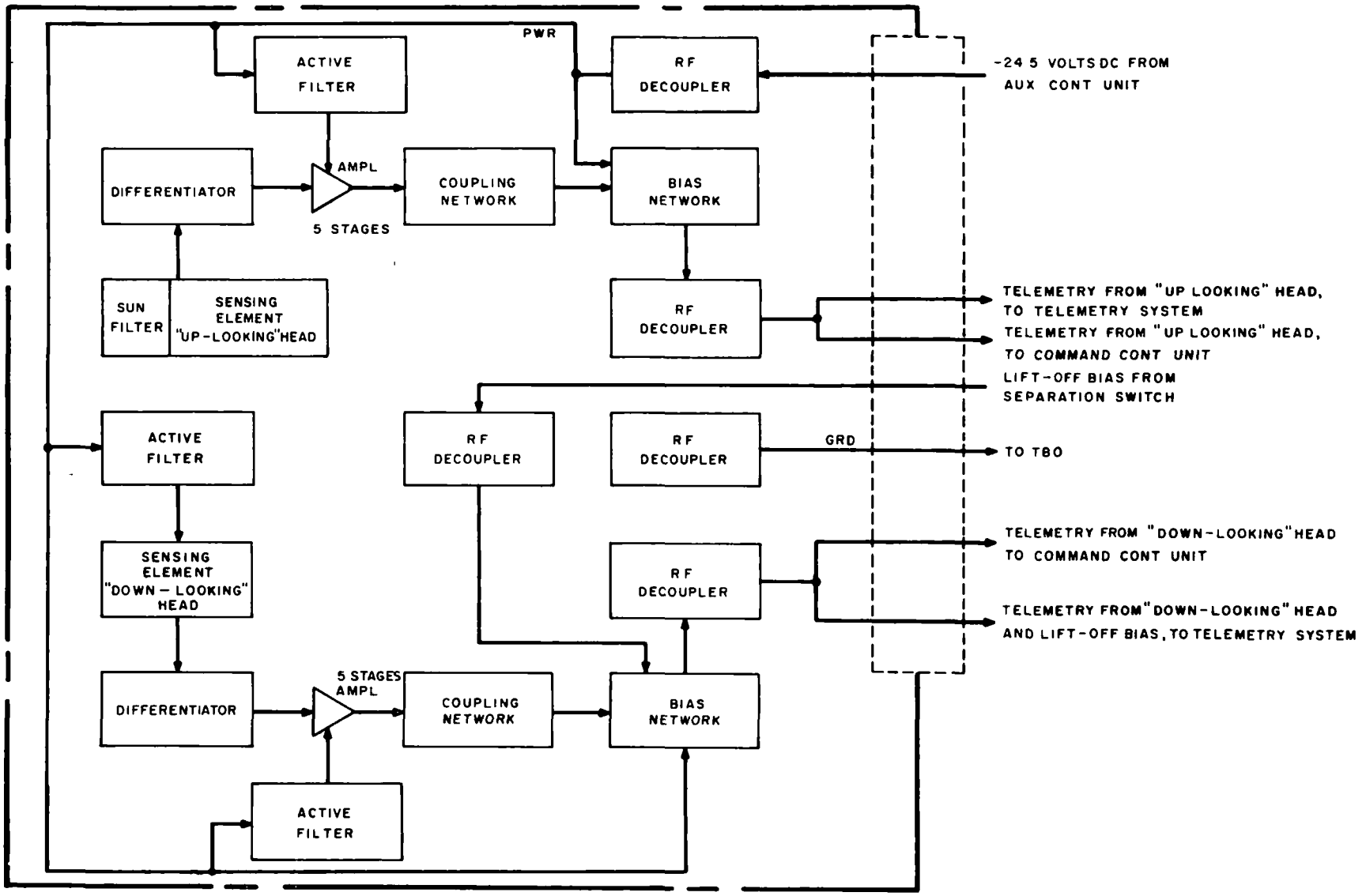


Figure II-20. Logic Diagram of V-Head Attitude Sensor

The outputs from the V-head attitude sensor are supplied to the primary SCO's in the telemetry system, and to the command control units for application to the secondary SCO's.

4. Solar-Aspect Indicator

a. General

The solar-aspect indicator utilizes a Gray-coded light mask which produces direct digital readings of the γ angle, i.e., the angle between the satellite-sun line and the satellite spin axis. The readings are presented as serial, Gray-coded words. The solar-aspect indicator has a 1-degree resolution over a range of 128 degrees. It consists of two separate subassemblies: the sensing element (the aspect indicator) and an electronics package. The sensing element consists of a 7-bit Gray-code reticle equipped with a small solar cell under each bit, and a double-slitted "command" reticle which also includes a solar cell. The solar-aspect indicator sensing element is shown in Figure II-21. The electronics package contains an amplifier for each bit, bi-stable multivibrators to establish thresholds and to convert the parallel input to a series

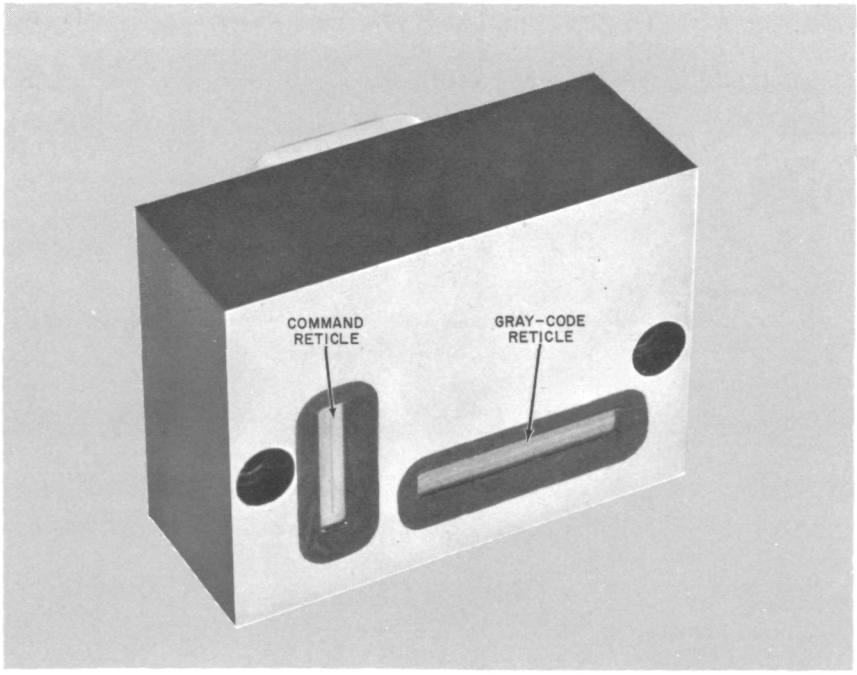


Figure II-21. Solar-Aspect Indicator, Sensing Element

output, and control circuits. The command reticle causes the angle determined by the Gray-coded reticle to be read out when the sun is in a plane defined by the spin axis and the satellite-sun line passing through the aspect eye.

b. Equipment Description

(1) Aspect Indicator

The reticle is a small oblong block of fused quartz with a slit centered along the top surface, and a light mask, arranged in a Gray-coded pattern, photographically applied to the bottom surface. Sunlight enters the indicator through the slit, casting a narrow band of illumination across the light mask. The light passing through the pattern generates a seven-bit word, coded to indicate the angle of incidence of the light on the mask. Gray code was chosen for the light-mask pattern to eliminate any errors in the measurement of the gamma angle caused by errors in the synchronization of the data bits. Such errors could be easily noted in the Gray code since only one data bit would change when the measured angle changes one degree. The spacing of the Gray-coded pattern is adjusted to correct for the nonlinear relationship between sun angle and distance from the center of the pattern, and is arranged to compensate for reduction in output due to cosine-law and reflection losses at extreme angles.

All of the critical mechanical tolerances in the aspect indicator are completely contained within the single block of quartz and are held to ±0.003-inch tolerance in the critical reticle dimensions. Since the accuracy of the aspect indicator is primarily a function of the constant ratio of the thickness to the width of the quartz block (with its integral components, the slit and reticle), temperature expansion or contraction of the block have no effect on accuracy.

Fused quartz is used since it does not fluoresce, discolor, or otherwise deteriorate when illuminated with ultraviolet or other forms of radiation encountered in space; its index of refraction varies less with temperature than other material; and it has a low linear-dimension temperature coefficient (1×10^{-6} °C).

Each of the seven bits on the Gray-coded reticle is superimposed on a photocell which detects the light (or notes the absence of light) in the aperture and produces a corresponding electrical output signal. P-on-N silicon solar cells are used as the photosensitive units. Although the output signal of this type of cell is relatively small (on the order of 40 to 50 microamperes) and therefore requires amplification, the cells can readily withstand the space environment and possess uniform outputs having a linear relationship to the incidence of light. The effects of ultraviolet and Van Allen radiation are minimized since the cells are substantially shielded by the fused quartz block.

The command reticle and its associated photocell is used as a trigger for the aspect indicator. The unit permits the sun angle to be read out only when the sun is contained in a plane at a right angle to the slit over the Gray-coded reticle. (This occurs once per satellite revolution.) The device consists of a reticle slitted on the top and bottom surfaces and mounted over a photocell The slits are oriented 90 degrees to the slit in the Gray-coded reticle.

(2) Electronics Package

A block diagram of the solar-aspect indicator is shown in Figure II-22. The output of each of the solar cells in the aspect indicator provides a maximum signal of approximately 50 microamperes into a low-impedance load. The signals from each bit are amplified by a one-stage, chopper-stabilized d-c amplifier and are applied to the first seven bits of an eight-bit shift register. Chopper transistors in the bit amplifiers are used to a-c couple the solar cell output to the amplifier, thereby preventing leakage in the solar cell due to the reversed-biased diode characteristics during high ambient-temperature conditions.

The command-eye produces a constant output at various gamma angles, and is connected to a d-c chopper which is driven by a 4-kc multivibrator. The chopped command-eye signal is amplified and coupled to a capacitive integrator through an emitter follower, which establishes the charge path for the integration of the chopped command-eye signal. The second output of the multivibrator is used to saturate a switch in the discharge path of the integrator circuit. The charge level of the integrator is dc-coupled through an emitter follower to the input of the Schmidt trigger. The combination of the relatively short RC time constant of the charge path, and the long RC time constant of the discharge path (switched in at a 4-kc rate) combine to produce the step pattern of the output waveform This output permits the Schmidt circuit to trigger at a discrete level, rather than on an undefined point on the command-eye wavefront.

When a command pulse is received from the command-eye amplifier, the negative going slope of the Schmidt-trigger output starts a 500-microsecond monostable multivibrator. The output of the multivibrator is inverted and applied to the chopper transistors in the bit amplifiers. The inverter output is also differentiated in the bit amplifiers and the negative half of the pulse is "AND'ed" with the collector of the bit amplifier in a manner that causes a "1" to be set into the register whenever the collector is above a predetermined level.

The compensated output of the bit amplifiers approaches a truncated triangular wave. Ideally, the transition from a "0" to a "1" should occur at the point where the light coming through the slit falls half on the opaque, and half on the transparent area of the Gray-coded reticle. The flip-flops in the register are designed to

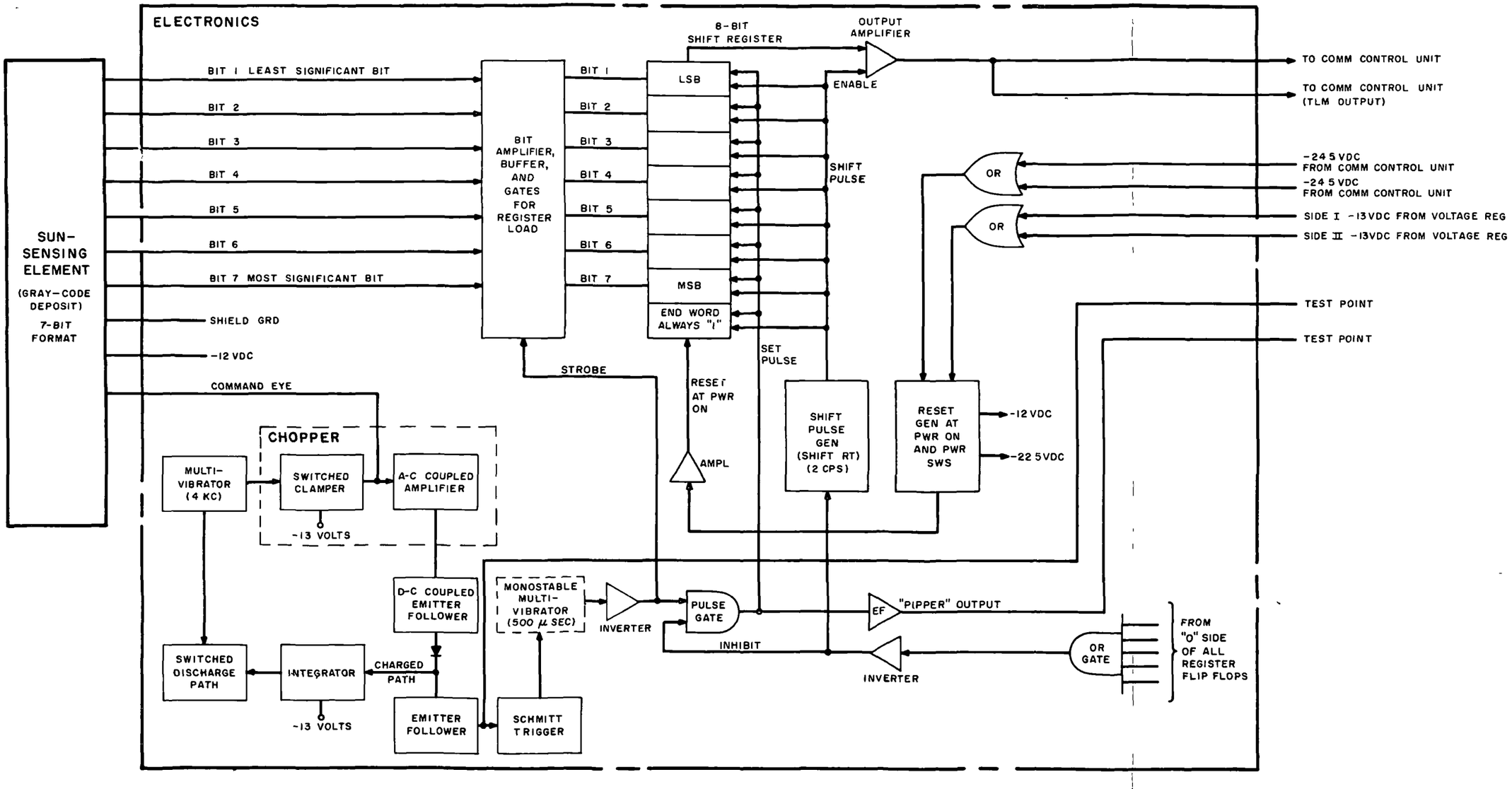


Figure II-22. Block Diagram of Solar-Aspect Indicator

trigger at the midpoint, setting at "1" for a signal above the midpoint and at "0" for below. A serial output is obtained by connecting the threshold flip-flops together in a shift register configuration. A shift oscillator is used to generate the shift pulses. The "0" side of all shift register flip-flops are applied to an OR circuit, whose output is used to enable the shift oscillator. A "1" is set into the "end-of-word" bit when the register is reset, therefore the shift oscillator will be enabled until this bit has been shifted out, after which the oscillator is disabled, ending the readout sequence.

On initial power turn-on, the integrator provides an output which is connected through diodes to the "0" side of the shift-register flip-flops and holds them OFF for 50 to 100 milliseconds. At the end of this period, a transistor switch is closed, disabling the "0" diodes until power is removed. The system will operate off a redundant set of d-c power supplies (-13 volts, and -24.5 volts) and can be switched to either set in the event of supply failure. (The -24.5 volt supply is turned on by a ground command; and the -13 volt supply is turned on by the solar-aspect indicator, itself.)

5. Testing

As noted previously, the components comprising the reference-indicator subsystem were items purchased by AED and, as such, were subjected to environmental qualification tests by the manufacturers prior to delivery.

F. POWER-SUPPLY SUBSYSTEM

1. General

The power-supply subsystem for the ESSA 1 satellite consists of an array of 9120 N-on-P silicon solar cells, an energy storage system containing 63 nickel-cadmium storage batteries, voltage regulation circuits, protection circuits, and telemetry sensing networks A special group of 60 cells, mounted on the top of the satellite hat, are included in the subsystem so that indications of the over-all condition of the solar cells can be telemetered to the CDA stations.

The solar cells are mounted on the top and sides of the satellite structure During orbital day, when the solar cells are illuminated by the sun, the output of the array is used as a primary power source for the satellite's electrical system; any excess power (power that is not needed by the satellite electronics) is used to charge the storage batteries. In cases where peak power requirements exceed the power output of the solar cells, the batteries automatically supply the power difference. During orbital night, the storage batteries, which have a total capacity of approximately 309 watt-hours, supply all the power required by the satellite. Precautions have

been taken to prevent circulating currents in the power source interconnections and to preclude the total loss of power in the event that a short circuit occurs in one of the battery cells.

The storage batteries are electrically connected in three independent groups, each of which is connected to the solar-cell array through its own charge-current controller. The charge controllers limit the maximum current into each battery string to a safe charge level. Excess array power is diverted through a bypass regulator to the main battery output bus. The bypass regulator also limits the maximum voltage excursion of the battery output bus so that input limits of the series-row voltage (-24.5 volts dc) are not exceeded. During orbital night, when the solar cells are passive, silicon diodes in each series row of solar cells prevent the storage batteries from discharging into the solar cells.

A similar function is performed by the diodes which are included in each series row of solar cells located on the lateral surface of the satellite. Because of the satellite rotation, each row is alternately illuminated and then darkened. The diodes prevent the darkened solar-cell rows from loading the illuminated rows. The storage batteries, in addition to providing power for the satellite subsystems, provide a relatively constant voltage across the solar cells, isolating them from variations in the electrical load.

Minus 24.5 volts and minus 13 volts are provided to supply the regulated d-c voltages required by the spacecraft. Minus 28 volts (unregulated) is supplied by the main battery output bus or the array bus via the bypass regulator, depending on which is more negative. Its limits are determined by the minimum battery bus power (minus 25 2 volts) and the maximum clamping level of the bypass regulator (minus 33 volts)

A block diagram of the power-supply subsystem is shown in Figure II-23.

2 Equipment Description

a. Solar-Cell Array

Electrical power for the satellite subsystems is generated by an array of 9120 N-on-P silicon solar cells mounted on the top and sides of the satellite structure Each cell is 1 x 2 centimeters in area, and has a transparent platelet of 0 006-inch, Corning, type-7940, fused-silica bonded to it to improve thermal emissivity. An antireflective coating which permits maximum light transmission in the 600- to 800-millimicron range (where the cells are most responsive) is vacuum-deposited on the upper (or outer) surface of each cell. A multilayer, sharp-cutoff, blue reflective coating is deposited on the lower or inner surface. This coating reflects all wavelengths lower than 400 millimicrons and transmits 90 to 95 percent of all longer

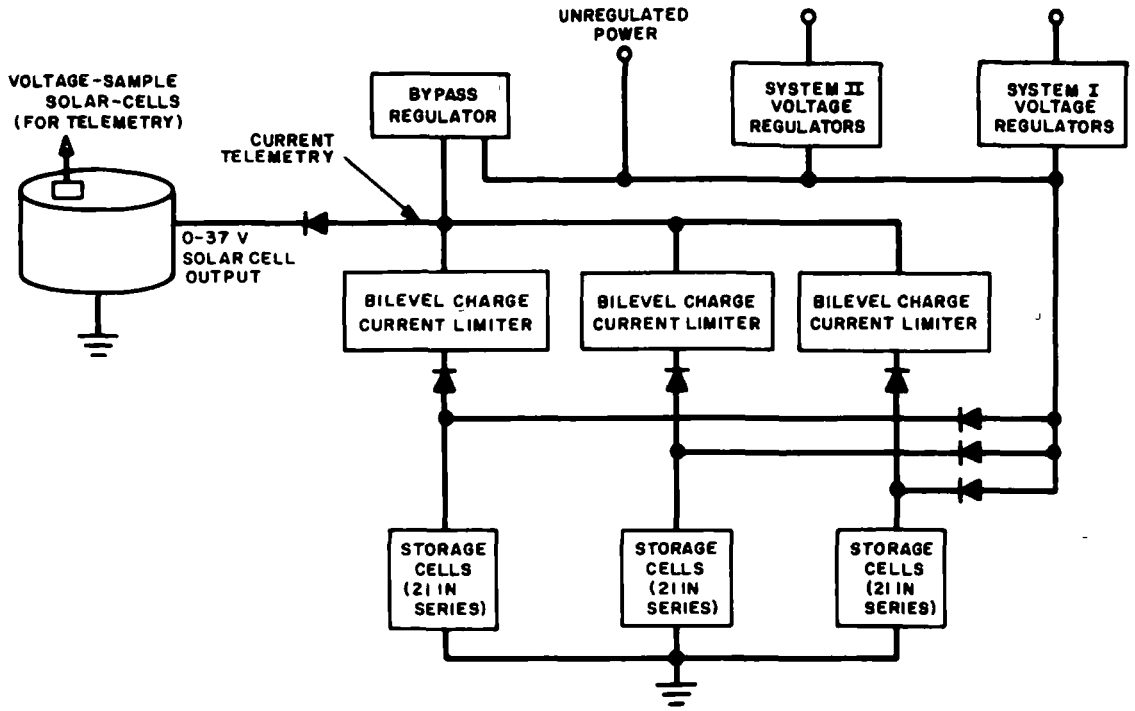


Figure II-23. Power-Supply Subsystem, Block Diagram

wavelengths up to and beyond the 1100-millimicron upper response limit of the cells. The platelet is bonded to each cell with a transparent epoxy adhesive.

The cells are assembled in shingle form, each shingle consisting of 5 cells connected in series. Each shingle has conversion efficiency of 9.0 percent at 1.95 volts and at a temperature of 27 ± 2°C. A group of sixteen shingles (i.e., 80 series-connected solar cells) are bonded to a flat epoxy-fiberglass board 3.400 inches wide by 7.572 inches long by 0.036 inch thick. The resultant module-board assembly weighs approximately 80 grams.

The modules are bonded directly to the top surface of the satellite and to the 18 side panels in parallel groups of four, with the zero (ground), one-quarter, one-half, three-quarters, and full voltage points connected in series. This method of interconnection tends to minimize effects of a shorted or open solar cell. The physical characteristics of the fully assembled solar array are as follows:

Area	No. of Cells	Active Area
Top	3520 (44 rows)	6 8 square feet
Sides	5600 (70 rows)	10.9 square feet

b. Solar-Cell Telemetry Patch

A group of solar cells mounted on the top of the hat is electrically isolated from the power supply and provided with a fixed resistive load that provides a d-c signal to the telemetry subsystem. When the top surface of the hat is illuminated, this group of solar cells provides an uninterrupted output voltage whose level is a function of solar-cell temperature and sun angle.

Prior to launching, the solar-cell telemetry patch was calibrated under a tungsten illumination source with an illumination intensity equivalent to the integrated solar-radiation intensity at satellite altitudes.

During the operational life of the satellite, the telemetered voltage output, V_T, of the solar-cell patch is periodically compared with the calibration voltage, V_C, for the specific solar-cell temperature and sun angle. This comparison, expressed by the ratio V_T/V_C, provides a measure of solar-cell performance and thereby, a means of checking solar-cell degradation due to radiation damage and micrometeorite bombardment.

c. Storage Cells and Battery Pack

The characteristics of the individual storage cells are as follows:

(1) Type of cell: Sintered plate glass-to-metal seal; hermetically sealed nickel-cadmium storage cell, F-size, with paper separators.

(2) Ampere-Hour Capacity:

(a) 3.9 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 25°C.

(b) 3.1 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 0°C.

(c) 2.7 ampere-hours at a 1.30 ampere discharge rate at a terminal voltage of 1.20 volts or greater at 40°C.

(3) Watt-hour capacity: 4.9 watt-hours at 25°C for discharge conditions listed in (a) above.

(4) Watt-hours per pound: 8.91 watt-hours at 25°C for discharge conditions listed in (a) above

(5) Maximum recharge rate: 0.600 ampere between 0°C and 40°C.

(6) Watt-hour efficiency: 70 percent.

(7) Physical Characteristics of Cell:

(a) Weight: 0.53 to 0.55 pound.

(b) Dimensions: Diameter: 1.3 inches,
Length: 3.5 inches,
Volume: 4.7 cubic inches.

The characteristics of the overall battery pack are as follows:

(1) Number of F-size cells: 63

(2) Number of parallel rows: 3

(3) Number of series cells per row: 21

(4) Total watt-hour capacity: 309 at discharge conditions defined in paragraph (2)(a) above.

(5) Total ampere-hour capacity: 1 7 at discharge conditions defined in paragraph (2)(a) above.

(6) Total weight of assembled battery including all packaging: 39 to 41 pounds

The following battery operational parameters are telemetered to the ground:

(1) The terminal voltage of each of the three rows of storage cells

(2) The voltage supplied to the input of one of the voltage regulators by the three rows of storage cells, each supplying power to the regulator through isolating diodes.

(3) The temperature of the battery pack.

(4) The output voltage of two minus 24.5 volt d-c voltage regulators and two minus 13.0 volt d-c regulators.

Each of these parameters is employed to determine whether the storage batteries and associated equipment are performing within the limits of the electrical design specifications.

d. Power Supply Protection Unit

(1) General

The power supply protection unit (PSPU) comprises (a) the array-current telemetry circuit, (b) the bi-level charge controllers, (c) the mounting assembly for the 10-ampere fuse wires, (d) the array-voltage telemetry circuit, (e) isolating diodes, (f) a bypass regulator, and (g) two pre-regulator circuits.

(2) Array-Current Telemetry Circuit

The array-current telemetry circuit permits an evaluation of the electrical performance of the solar-cell array under orbital conditions. The magnitude of current flowing in the solar-cell array is measured, and the measured value is converted to a signal voltage suitable for transmission to the ground by means of the telemetry system. The maximum output signal of the array-current telemetry circuit, with 2.5 amperes flowing through the array current sensing resistors, is set at -5 volts dc. The circuit is normally in the OFF state, and is commanded ON by a -24.5 volts dc signal.

Figure II-24 is a schematic diagram of the array-current telemetry circuit. Referring to this diagram, the circuit can be divided into four major sections as follows:

- Current-sensing resistors R1 and R2.
- A d-c amplifier consisting of Q1-A, R3, R8, R9, CR2 and R7.
- A temperature-compensating bias source for the amplifier consisting of Q1-B, R4, R6, CR1, and R5.
- A combination switch and constant current source for the bias network consisting of Q2, R10, R11, R12, C1, and CR3.

A detailed description of the array-current telemetry circuit is presented in the TIROS IX Final Engineering Report.

(3) Bi-Level Charge Controllers

The bi-level charge controllers are included in the power-supply subsystem to permit a control of the maximum charge current supplied to the battery strings from the solar-cell array. The three 21-cell battery strings comprising the battery pack are independently charged by the solar-cell array during the daylight

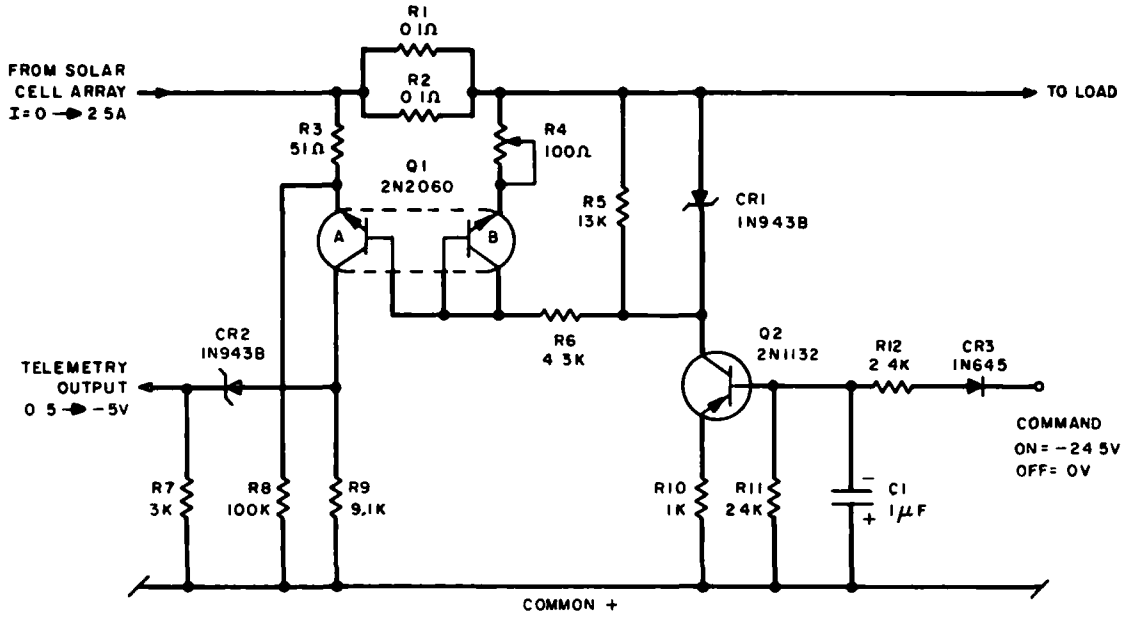


Figure II-24. Array-Current Telemetry Circuit, Schematic Diagram

portion of each orbit. Charge current to each battery string passes through a bi-level charge controller which, under normal conditions, serves to limit the maximum charge current to a nominal 440 milliamperes.

The charge current to the battery strings can also be adjusted to a 550-milliampere level, by means of a command from a CDA station. This capability can be utilized in the case of a loss of a battery string, with the higher, less conservative charge rate being used to maintain energy balance. In addition, it can be utilized to compensate, to a limited degree, for the decrease in charging efficiency of the nickel-cadmium cells due to aging. (Such a decrease would reduce the energy available during the dark portion of an orbit.)

The bi-level charge controllers are only capable of limiting the maximum charge current to the battery. Minimum charge current at any time during the daylight portion of the orbit will be determined by the battery terminal voltage, array bus voltage, and the load profile. Figure II-25 presents a schematic diagram of the bi-level charge controller; further detail on this unit can be found in the TIROS IX Final Engineering Report.

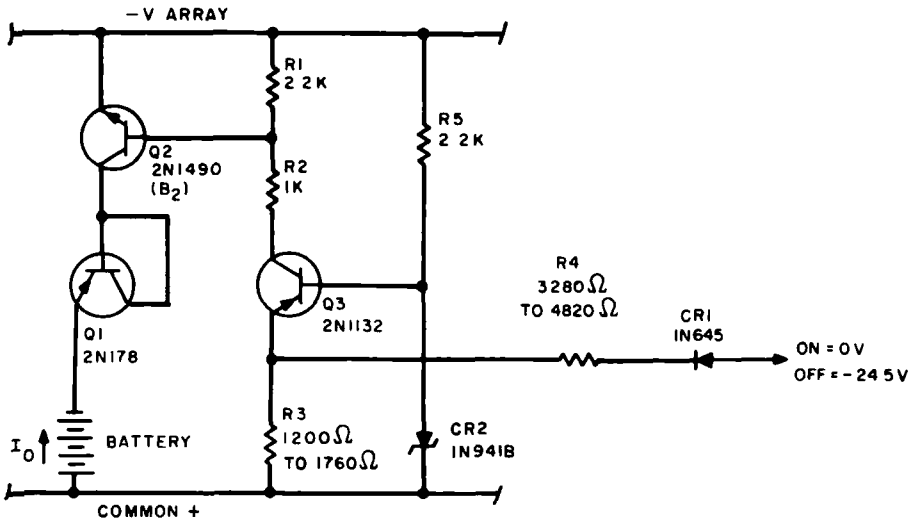


Figure II-25. Bi-Level Charge Controller, Schematic Diagram

(4) Fuse-Wire Mountings

Five 10-ampere fuse wires are connected between the unregulated bus and the major subsystem elements, and two 1.2-ampere fuse wires are placed between the unregulated bus and the two pre-regulators

The five 10-ampere fuses were wired in a fiberglass fuse housing which completely encloses the individual fuses and is mounted on a bracket on top of the PSPU.

(5) Array-Voltage Telemetry Circuit

The array-voltage telemetry circuit is a precision, resistive divider network connected directly across the output of the solar array, and provides telemetry level signals directly proportional to the array output.

(6) Isolating Diodes

Six high-current transistors, connected as diodes, are used to isolate the charging and load-delivering paths of each 21-cell battery string. These diodes prevent (1) battery discharge into the satellite loads during the daytime (except during periods of high-current requirements), and (2) battery discharge into the quiescent solar-cell array during the orbital night. The units also prevent circulating currents

between the parallel-connected battery strings and, in the event of failure in one string, preclude total loss of power by isolating the defective battery string.

(7) Bypass Regulator

Under normal conditions, the solar-cell array can deliver 2.2 amperes to the satellite. Of this total, whatever is required for direct use by the satellite loads will be supplied, and the remainder, to a maximum of 1.32 amperes with the low-level charge rate and 1.65 amperes with the high-level charge rate, will be supplied to the batteries by means of the bi-level charge controllers.

A simple two-transistor bypass regulator prevents the voltage level of the unregulated power bus from exceeding 32 volts. When the input drops below 32 volts, the bypass regulator functions as a diode with a forward drop of approximately 1.0 volt and passes the current to the load without regulation.

(8) Pre-Regulators

Two pre-regulators, connected across the unregulated bus, clamp their outputs to -16 ± 1 volts dc, providing pre-regulated inputs to the two -13 volt dc series voltage regulators

e. Voltage Regulators

The ESSA 1 voltage regulators consist of two assemblies, each containing a -24.5 volt dc voltage regulator, in one box, and, a -13 volt dc unit in a second, independent box mounted on top of the first.

A schematic diagram of the -24.5 volt regulator is shown in Figure II-26 Transistors Q5 and Q6 form a differential amplifier which compares voltage deviations across the load with a voltage reference zener diode CR1. A balanced output is directly coupled to the bases of Q3 and Q4, a second differential amplifier connected in complementary symmetry. A stage of current gain, Q2, provides the necessary drive to the series pass transistor Q1 to support the required load current.

A schematic diagram of the -13 volt regulator is shown in Figure II-27. The -13 volt regulator is composed of transistors, Q101 through Q105, and associated components. The operation of the regulator is much the same as the -24.5 volt regulator. There are, however, several differences. The -13 volt regulator was designed to carry loads up to 0.5 amperes whereas the -24.5 volt regulator is intended for loads up to 5.0 amperes. As a result of this difference, the -13 volt regulator need not

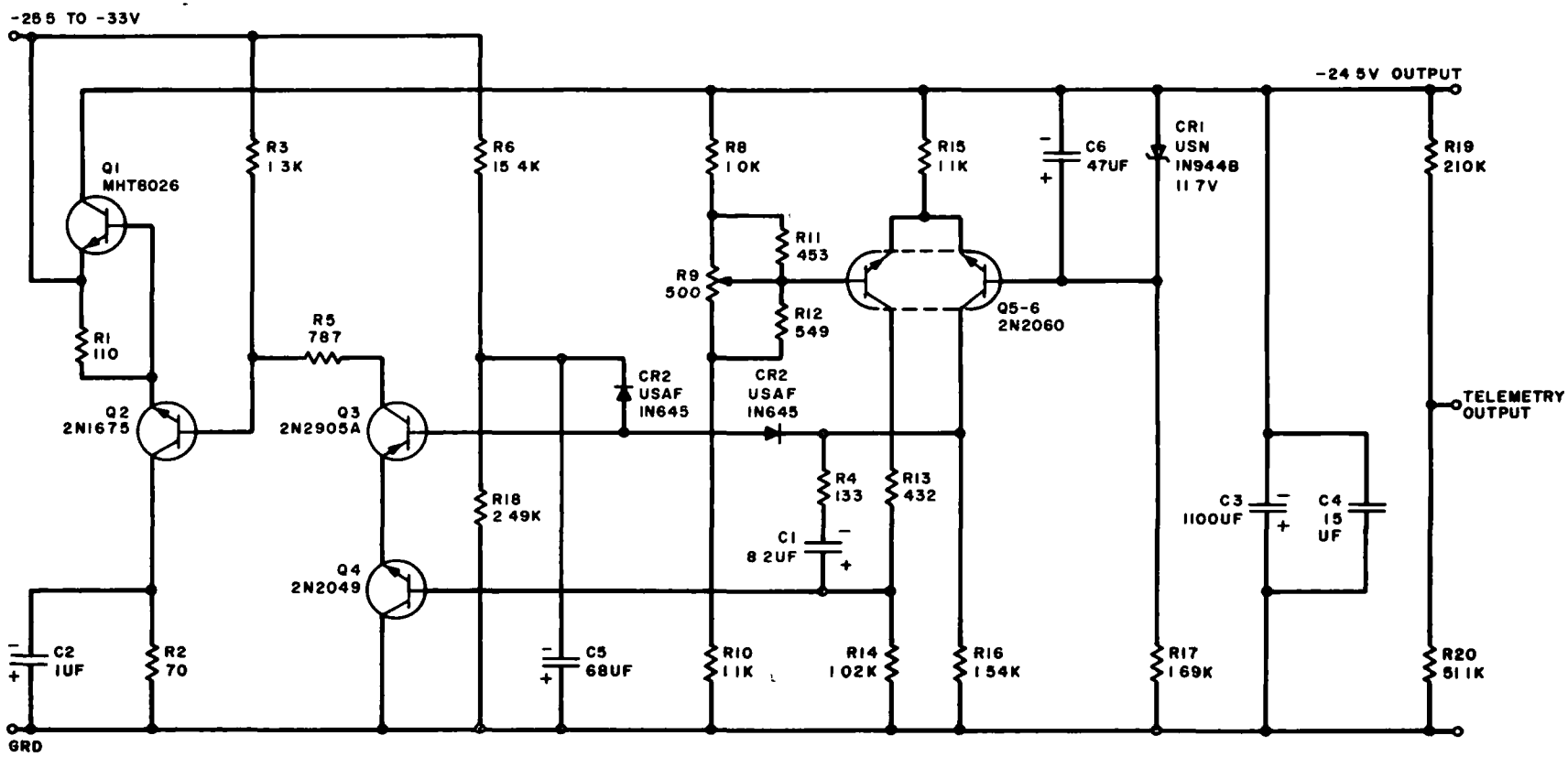


Figure II-26. Minus 24.5-Volt Regulator, Schematic Diagram

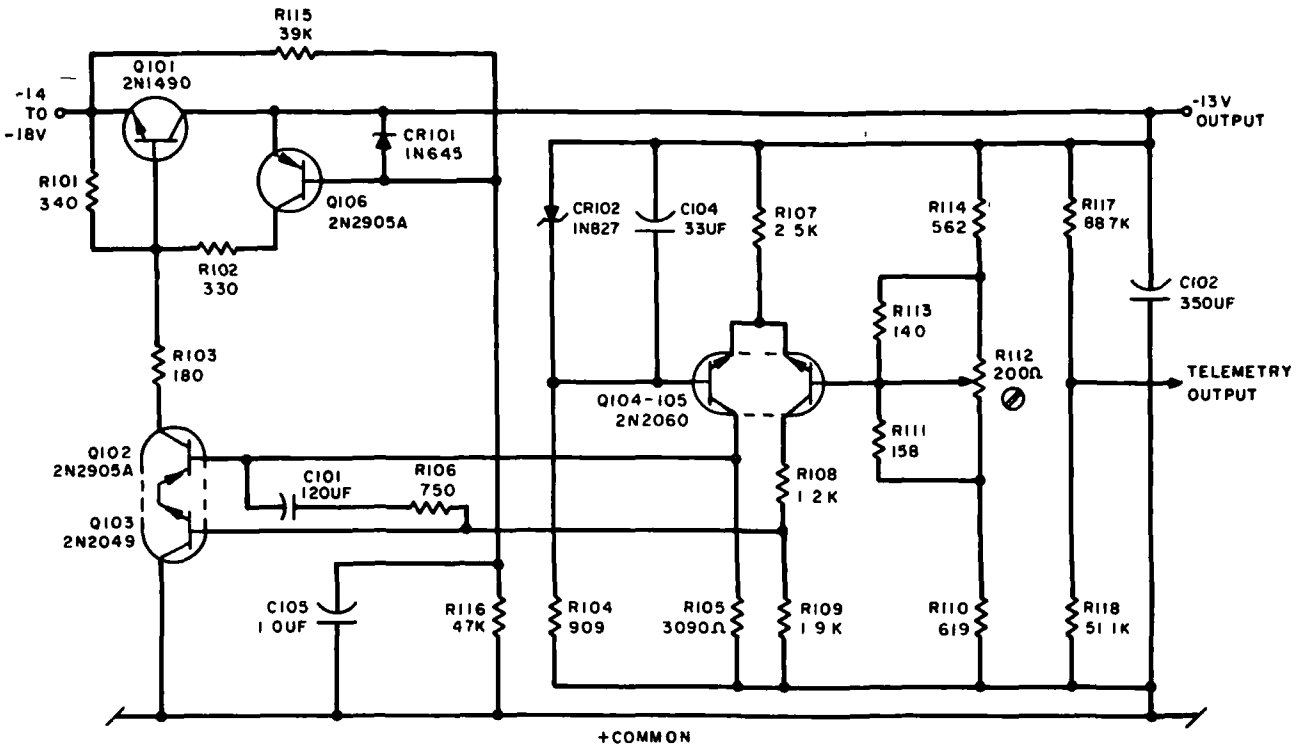


Figure II-27. Minus 13-Volt Regulator, Schematic Diagram

have as large a series regulating circuit as the other; in addition, the regulator loop gain does not have to be as high.

Further detail on the -13 and -24.5 volt regulators is presented in the TIROS IX Final Engineering Report.

3. Testing

a. Solar-Cell Array

An outdoor test of the ESSA 1 solar-cell array was performed in November 1964 During this test the power output was measured on a panel-by-panel basis. Though no compensations were made for cell temperatures and absolute power input, the I_{sc} values obtained were still in excess of requirements. On December 2, 1965, a second outdoor test was conducted to measure the continuity of the side panels, and an indoor test was conducted to measure the continuity of the top-hat solar-cell array. Satisfactory results were evident throughout both tests.

b. Battery Pack

The ESSA 1 flight-model battery pack, Serial No. 3-721-08, was electrically tested on two occasions in April 1965 with satisfactory results. The pack was placed in storage at 0° C in July 1965, and maintained at that temperature, except for periodic capacity tests at 25 °C. In November 1965, upon receipt of the NASA directive to remove spacecraft OT-3 from the "hold" condition, the Serial No. 3-721-08 battery pack was removed from storage. The pack was integrated on spacecraft OT-3 on November 11, 1965.

The test results obtained in the qualification testing of the ESSA 1 flight-model battery pack are presented in Table II-4. Test specifications required that (1) the total row voltage after charge at 1 amp for 6 hours should not be more than 31.1 volts and (2) the total row voltage after discharge at 1.3 amps for 3 hours should not be less than 25.2 volts. It should be noted that in no case did the ESSA 1 battery pack fail to meet these specifications.

c. Power Supply Protection Unit

The power supply protection unit, Serial No. 3-723-07, was successfully subjected to vibration testing on September, 1964, and successfully completed thermal-vacuum testing on September 3.

d. Voltage Regulator Assemblies

The ESSA 1 voltage regulator assemblies, Serial Nos. 3-727-04 and -05, each comprising a -13 volt dc voltage regulator and a -24.5 volt unit, were successfully subjected to vibration testing on August 17, 1964, and both assemblies successfully completed thermal-vacuum testing on August 26. During thermal-vacuum testing, the stabilization and regulation characteristics of each assembly were measured

A modification made to the spacecraft "I" voltage regulator assemblies during the latter half of September 1964 was also made to the spacecraft OT-3 units This modification consisted of removing a 0.56-microfarad high-frequency bypass capacitor from the output of the -13 volt voltage regulator so as to improve the output impedance characteristics of the unit at low temperatures. It had developed that the wiring harness within the satellite negated any contributions that the capacitor made to the high-frequency characteristics of the voltage regulator. After the modification had been implemented, all characteristics of the units were re-examined under the required thermal conditions to verify that no changes had occurred.

TABLE II-4. RESULTS OF QUALIFICATION TESTING OF BATTERY PACK, SERIAL NO. 3-721-08

Results of Initial Tests Completed 4/1/65	ROW X (Volts)	ROW Y (Volts)	ROW Z (Volts)
Total row voltage after charge @ 1 amp for 6 hours	30.45	30.60	30.54
Total row voltage after discharge @ 1.3 amps for 3 hours	25.64	25.64	25.75
Results of Tests Conducted 4/9/65			
Total row voltage after charge @ 1 amp for 6 hours	30 30	30 50	30.43
Total row voltage after discharge @ 1.3 amps for 3 hours	25.63	25.70	25.77
Results of Tests Conducted 8/12/65			
Total row voltage after charge @ 1 amp for 6 hours	30.63	30.38	30.81
Total row voltage after discharge @ 1.3 amps for 3 hours	25.79	25.67	25.84
Results of Tests Conducted 9/30/65			
Total row voltage after charge @ 1 amp for 6 hours	30.38	30.42	30.38
Total row voltage after discharge @ 1.3 amps for 3 hours	25.70	25.45	25.74
Results of Tests Conducted 11/15/65			
Total row voltage after charge @ 1 amp for 6 hours	30.23	30 15	30.16
Total row voltage after discharge @ 1.3 amps for 3 hours	25.62	25.39	25.47

G. ANTENNA SUBSYSTEM

1. General

The ESSA 1 antenna subsystem provides for the reception of command signals from the CDA ground stations and for the simultaneous radiation of energy from three of the four separate satellite transmitters (two beacon transmitters and either one of the two TV transmitters). The subsystem also provides for coupling and matching the receivers and transmitters to the antennas and isolating the three active transmitters and the two command receivers.

The subsystem consists of (1) a single dipole receiving antenna for the command receivers, (2) two crossed-dipole transmitting antennas, (3) the associated RF matching and coupling network for the transmitters, (4) bandpass filters and attenuators to reduce beacon spurious output and intermodulation between beacon transmitters, (5) a notch filter for further isolating the command receivers from the beacon transmitters, and (6) a 4-port hybrid to equalize the strength of the signals received by the two command receivers. An interconnection diagram of the ESSA 1 RF system is shown in Figure II-28.

2. Equipment Description

a. Receiving Antenna

The receiving antenna is a separate 1/4-wavelength dipole at the command frequency. It is positioned in the neutral plane of the crossed transmitting dipoles. This positioning causes an approximate 45-db attenuation to exist between the transmitting and receiving antenna terminals. The receiving antenna is vertically mounted on top of the satellite at the spin axis and is coupled to the two receivers through a 1/2-wavelength transmission line, a four-way cross adapter, and 3/8-wavelength transmission lines.

b. Transmitting Antenna

The two crossed-dipole transmitting antennas form a composite antenna operating at both the beacon and the TV frequencies. It consists of four elements mounted to the satellite baseplate. Each element is a coaxial structure with rods extending through canted sleeves. The total element length is approximately 32 5 inches which is equivalent to 0.36 wavelength at the 136-Mc beacon frequency. At the TV frequency (235.0 Mc), the rod extensions are isolated from the coaxial sleeves by a short-circuited stub within each coaxial structure. Thus, at that frequency, the element length is that of the sleeves, which extend 12 inches from the

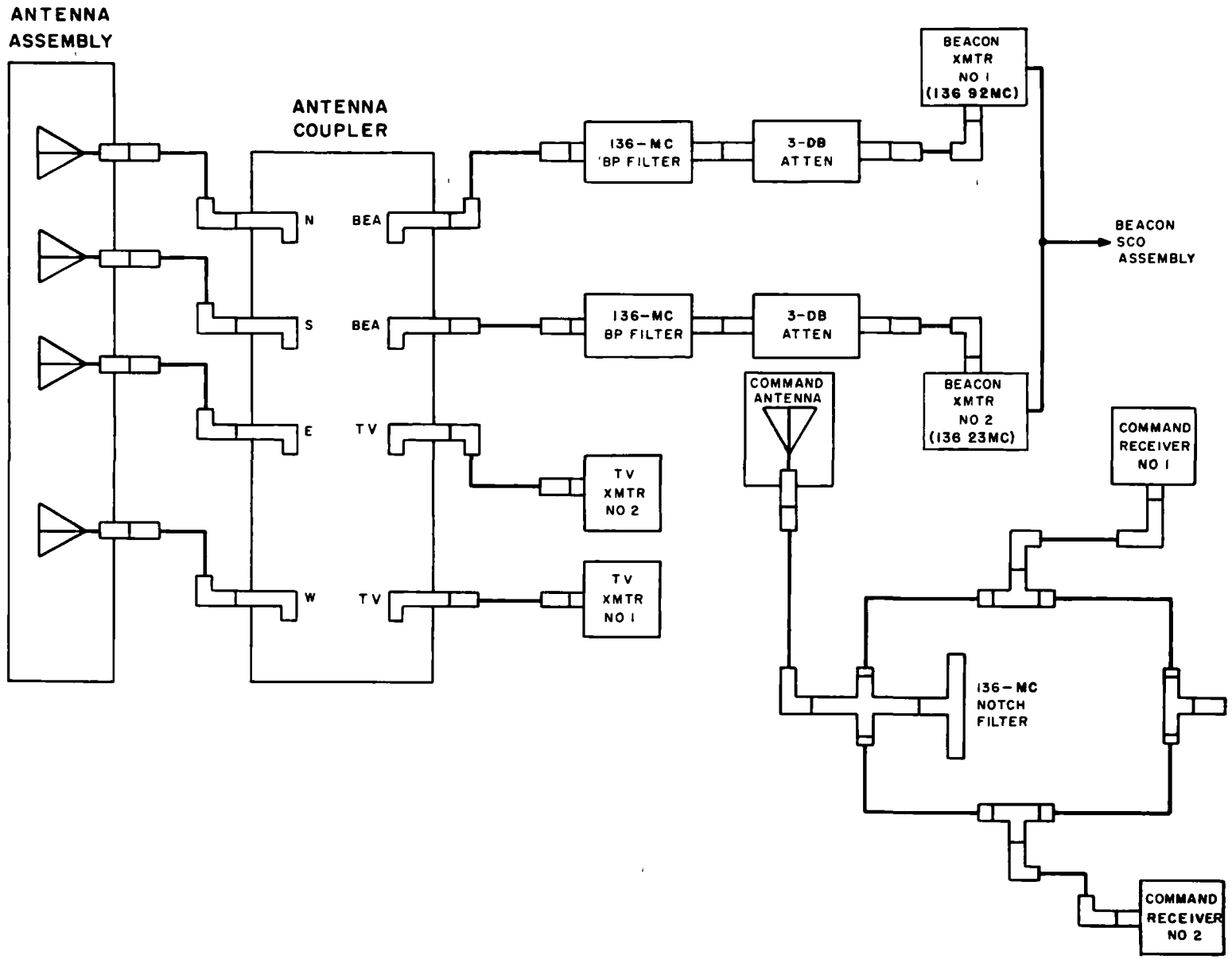


Figure II-28. RF System Interconnection Diagram

drive end of the dipole. This length is equivalent to 0.25 wavelength at 235 Mc. The dipoles are fed in quadrature to achieve circular polarization.

c. Matching and Coupling Network

The ESSA 1 matching and coupling network is the same as that used in TIROS IX. The network couples the three transmitters to the radiating elements, provides an impedance match for this coupling, minimizes interaction and feedback between the transmitters, and effects circular polarization by exciting the antenna elements in phase quadrature.

Figure II-29 shows the schematic diagram of the matching and coupling network. The network is composed of two similar sections, each consisting of a diplexer and two baluns. The RF transmitters (shown as a-c generators in Figure II-29) are connected symmetrically across the diplexer inputs, while the matched loads are driven from the output. The TV transmitters are connected directly to the 235-Mc diplexer inputs. However, on ESSA 1, the connections between the beacon transmitters and the 137-Mc diplexer inputs were modified, as detailed later in this subsection, to include a 136-Mc bandpass filter and a 3-db attenuator pad between each beacon transmitter and the diplexer input. The currents in the two loads are phase-displaced by 180 degrees. Theoretically, the two transmitters in each section are completely isolated, resulting in a perfect load match.

The two baluns in each section are essentially delay lines that selectively phase the RF currents in the dipoles To achieve circular polarization, the two line lengths between the diplexer outputs and their respective baluns differ in electrical length by 90 degrees.

d. Notch Filter

A notch filter is connected to one input of the coaxial adapter to provide isolation of the command receivers from the beacon transmissions. The filter is identical to the one used on TIROS IX, and is in the form of a shunt stub-line placed across the antenna line. The stub is open-circuited and one-quarter wavelength at the frequency to be rejected, thereby providing a high conductance at the rejected frequency, but an extremely low conductance at the command-receiver frequency. The achievement of a high conductance at the rejection frequency in the 137-Mc band and a negligible conductance at the relatively close command frequency required a line having both a high Q and high characteristic impedance.

The RF filter used to satisfy these requirements utilizes a coaxial transmission line having a helical inner conductor, the diameter of which is approximately half the inner diameter of the outer conductor.

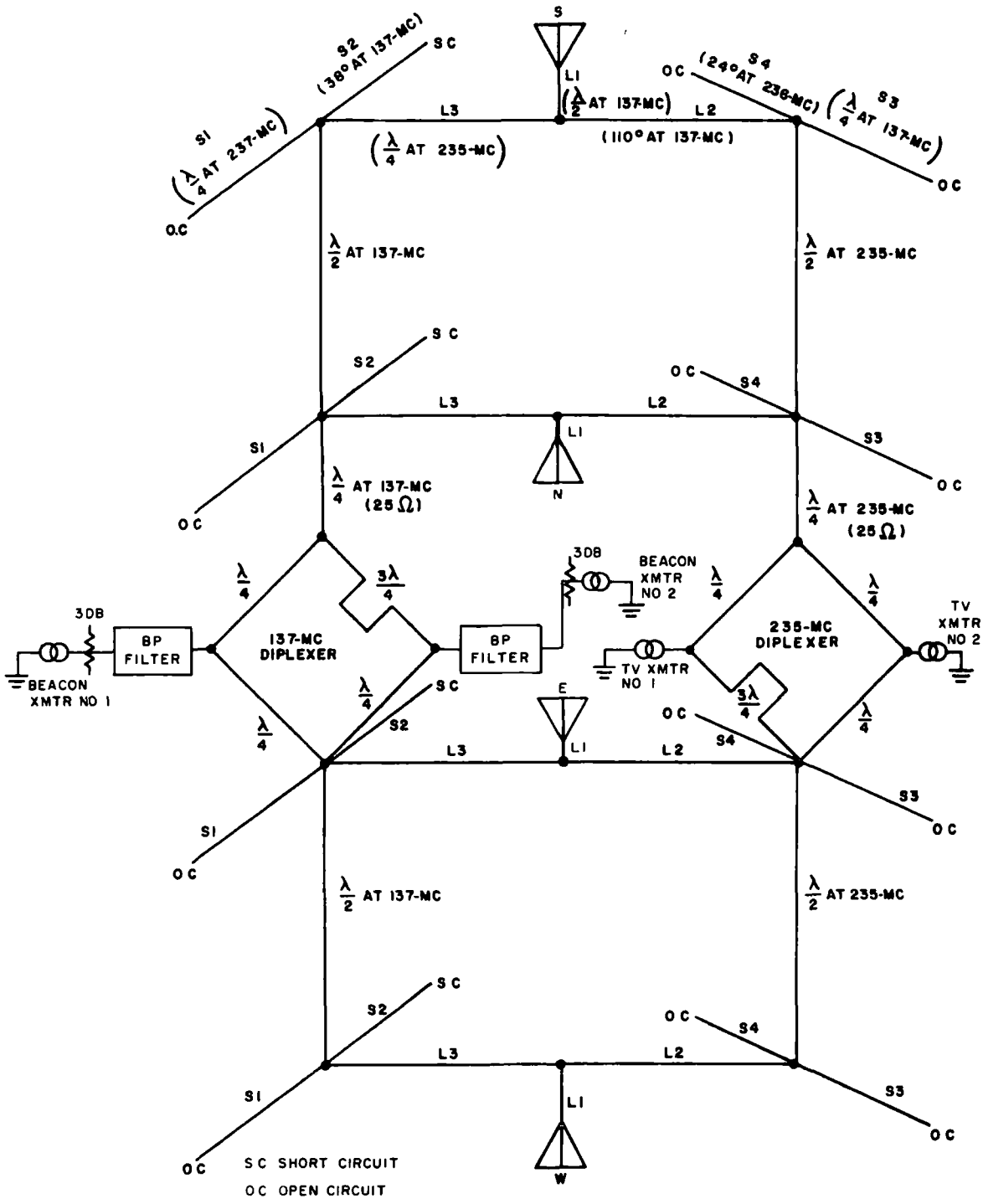


Figure II-29. Matching and Coupling Network, Block Diagram

The ESSA 1 notch filter was tuned for maximum isolation at 136.57 Mc, i.e., the median of the beacon frequencies.

e. 4-Port Hybrid

During qualification testing of spacecraft OT-3, a defective cable in the command receiver harness was detected. It was decided at that time that rather than simply replacing the cable it would be best to change the entire harness to utilize a 4-port hybrid to achieve the additional advantages of equalizing the signal strength received by each of the command receivers and eliminating the possibility of receiver oscillations. An admittance matrix analysis of the 4-port hybrid is presented in Appendix A.

f. Attenuators and Bandpass Filters

As described below, as a result of RF systems tests performed to assess the compatibility of the spacecraft and the instrumented third-stage launch vehicle, a 3-db attenuator and a 136-Mc bandpass filter were installed in series with the output of each beacon transmitter. The objectives of this modification were to increase the isolation between beacons and reduce spurious outputs.

3. RF Systems Tests

On November 18 and 19, 1965, an RF systems test of spacecraft OT-3 was performed to assess the compatibility of the spacecraft and the instrumented X-258 third-stage launch vehicle. For this test, the spacecraft was mounted on a simulated third-stage launch vehicle, thereby approximating the conditions that exist during the RF systems tests conducted at the launch site during prelaunch operations. This test was motivated, in part, by studies of TIROS X evaluation data and, in particular, by a study of command sensitivity of the satellite as a function of beacon transmitter radiation. In addition, simulation tests conducted as part of the TIROS X evaluation indicated that the isolation between beacon transmitters must be at least 16 db to prevent excessive intermodulation of the beacon transmissions, which would be capable of producing spurious signals within the command receiver passband.

In the tests performed on November 18 and 19, difficulty was experienced in commanding the spacecraft while it was attached to the simulated third-stage vehicle, indicating command-receiver desensitization. When the two structures were separated by as little as a few inches, must less desensitization was indicated.

The spacecraft RF system was originally designed so that in free space the antennas presented a matched and balanced load to the coupling unit or diplexer.

Under free-space conditions, the diplexer provided a 50-ohm termination impedance to each beacon transmitter and an isolation between the beacon transmitters of 26 db.

When the spacecraft is attached to the simulated third-stage spacecraft, the RF system becomes detuned and the beacon transmitters then operate into a mismatch. The results are a VSWR of 1.4 (nominal). In addition, the isolation between the command (receiving) antenna and the transmitting array is reduced for signals at the command frequency.

As a result of these changes in the RF operating environment of the beacons, distortion and spurious frequencies were noted. The results of the system test for spurious signal generation by the beacon transmitters are given in Table II-5. It was later found that the beacon modulation percentage was somewhat high, tending to further enhance the generation of spurious frequencies.

Of the many spurious frequencies detected, two were of particular significance: 147.5 Mc and 148.3 Mc. These frequencies were traced to a crystal harmonic (each is 13/12 times a beacon frequency). A series of spare beacons were tesed; the spurious generation tendency varied widely with signals (near the command frequency) with power levels as high as -10 dbm. The best of the units tested were selected for use on spacecraft OT-3, based on the criteria detailed in Appendix B.

Measurements were also made to determine the effect of better isolation between the beacons. It was found that by inserting a 3 db attenuator in series with each beacon output the levels of the spurious outputs were substantially reduced. These attenuators also lower the VSWR to near unity. While these attenuators removed the gross effects of the X-258 upon the beacons, it was felt that further improvements were needed in case spurious frequencies were generated by other means. Accordingly, 136 Mc bandpass filters were procured, and a filter was installed in series with each beacon output. The attenuator pads and filters afforded about 5-db loss in the 136 Mc beacon band, and over 35 db on the region of 148 Mc. These system modifications were made and effectively solved the problems of the isolation and VSWR.

In the course of the investigations, the percent modulation of the beacons was optimized to obtain minimum crosstalk, distortion, and tendency to produce spurious RF frequencies. The modulation levels established were as follows: high-frequency beacon—80 to 85 percent; low-frequency beacon—70 to 75 percent.

Analyses of the effects of modulation index on the signal-to-noise ratio at the CDA station receiver output and FM demodulator output and on the spurious RF signal levels aboard the spacecraft were carried out in conjunction with this system test; the results of these analyses are presented in Appendix B of this report. The path loss and margin calculation for the modified RF system and the calculation of the worst-case spurious input to the command receiver for the modified RF system are also included in this appendix.

TABLE II-5. BEACON TELEMETRY DATA CHANNEL SPURIOUS SIGNAL STRENGTHS AT DIFFERENT BEACON MODULATION LEVELS FOR SPACECRAFT OT-3

Percent Modulation	Low Beacon*		High Beacon*	
	Chan. 5	Chan. 7	Chan. 5	Chan. 7
	fs: 1200 cps fi: 1250 cps	fs: 2450 cps fi: 2400 cps	fs: 1200 cps fi: 1250 cps	fs: 2450 cps fi: 2400 cps
50	X	X	-23db	-28db
60	-33db	-28db	-25db	-30db
70	-24db	-28db	-28db	-32db
80	-22db	-26db	-34db	-30db
90	-20db	-23db	-27db	-32db

fs is the frequency of the desired signal

fi is the frequency of the strongest spurious signal

*Figures expressed in db are amplitudes of spurious signals (fi) relative to desired signals (fs).

4. Testing

On December 11, 1963, both the 136 Mc and the 235 Mc sections of the matching and coupling network, Serial No. 3-424-04, were successfully subjected to electrical acceptance tests. The sections were then combined and assembled with the transmitting antenna, Serial No. 3-426-04. On September 1, 1964, the electrical acceptance testing of the antenna, i.e., the tuning of the antenna, was successfully performed. Electrical acceptance testing of the integrated matching and coupling network was performed on September 3, with satisfactory results.

Electrical acceptance tests of the notch filter were successfully completed on December 8, 1964. The response characteristics of the filter, as measured during these tests, are shown in Figure II-30.

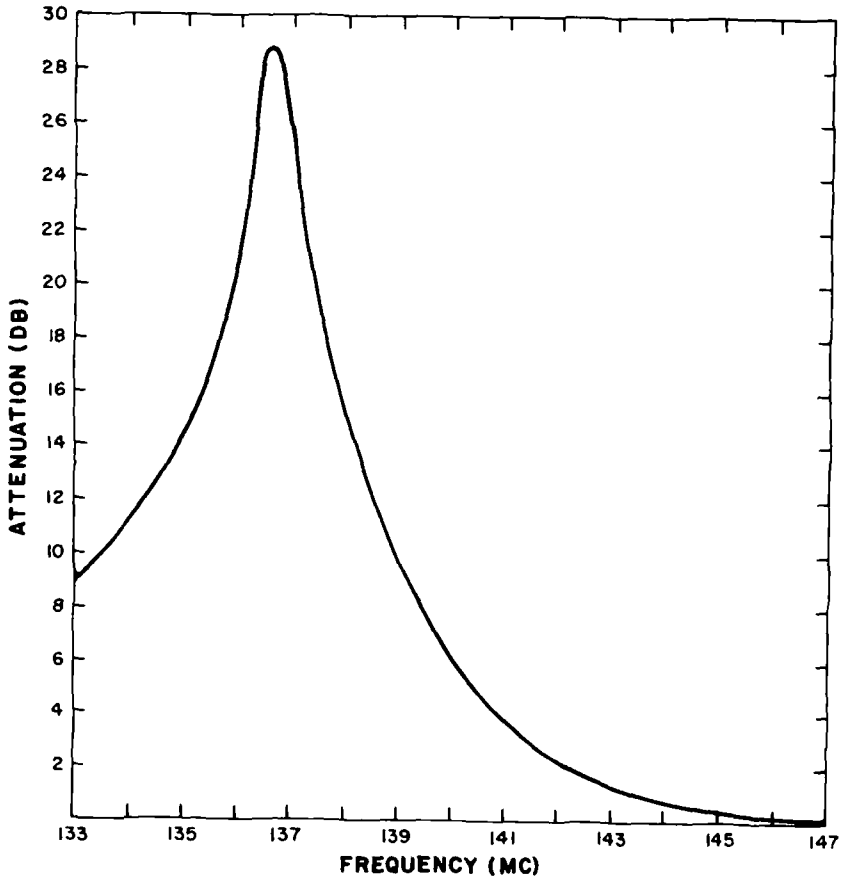


Figure II-30. Response Characteristics of ESSA 1 Notch Filter

H. DYNAMICS-CONTROL SUBSYSTEM

1. General

The dynamics-control subsystem on ESSA 1 is identical to that used in TIROS IX. It is comprised of the following units:

- precession dampers,
- despin device,
- spin-up rockets,
- QOMAC (Quarter Orbit Magnetic Attitude Control) coil,

- MBC (Magnetic Bias Control) coil, and
- MASC (Magnetic Spin Control) coil.

The two redundant Dycon (dynamics control) units provide a means of programming operation of the AOMAC and MASC coils; the Dycon units also provide a back-up means of camera triggering based on the spin-synchronized principle.

The Dycon units are logically a portion of the programming and control subsystem, which is detailed in Volume II (the classified supplement) of this report: however, since the operation of the Dycon units is so closely involved with the dynamics-control subsystem, the description of the Dycons is included in this section of the report.

The QOMAC coil is programmed, by means of the Dycon, to accomplish the wheel maneuver in the early orbits of the satellite. The QOMAC coil is then used for maintaining the satellite in the wheel attitude.

The MBC coil is used for nulling the residual magnetic dipole of the satellite and for producing an additional dipole which will reduce the number of QOMAC torquing cycles necessary for maintaining the satellite in the proper attitude. The MASC coil is used to decrease or increase the satellite's spin rate and offers finer adjustment in increasing the spin rate than is available with the spin-up rockets. The QOMAC and MBC coils are wound on the same frame and mounted inside the spacecraft hat, orthogonal to the MASC coil. The associated magnetic dipoles are shown in Figure II-31.

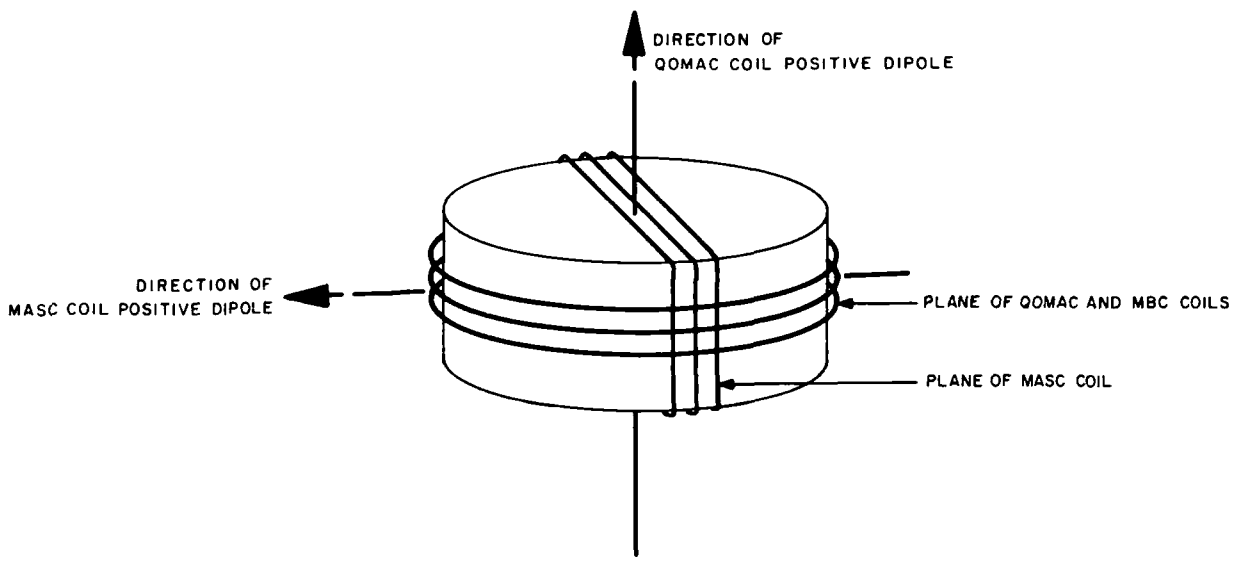


Figure II-31. Magnetic Dipoles Associated with QOMAC and MBC Coils

2. Precession-Damping Devices

The initial wobble of the satellite, caused by precession or nutation after release from the third-stage rocket, is damped-out rapidly by two tuned-energy-absorption-masses (TEAM) that oppose the forces that tend to oscillate the satellite body. The precession-damping devices are shown in Figure II-32. Two similar mechanisms are installed vertically along the inside wall of the satellite hat, 180 degrees apart. Each mechanism weighs a little under one pound; each mass, approximately 3 ounces. During ascent, each travelling mass is restrained by a mechanical gate. Upon separation of the satellite from the third stage of the launch vehicle, the gates are opened by automatic firing of squibs, permitting the masses to roll freely along the rods. Manual firing of squibs from the ground station, utilizing switch contacts on the rocket-firing switch in the satellite, is also provided as a back-up control measure. The device is tuned to the natural precession frequency of the satellite and rapidly absorbs the energy causing the wobble, dissipating this energy in the form of heat. Upon stabilization, the masses rest at or near the center of the rods.

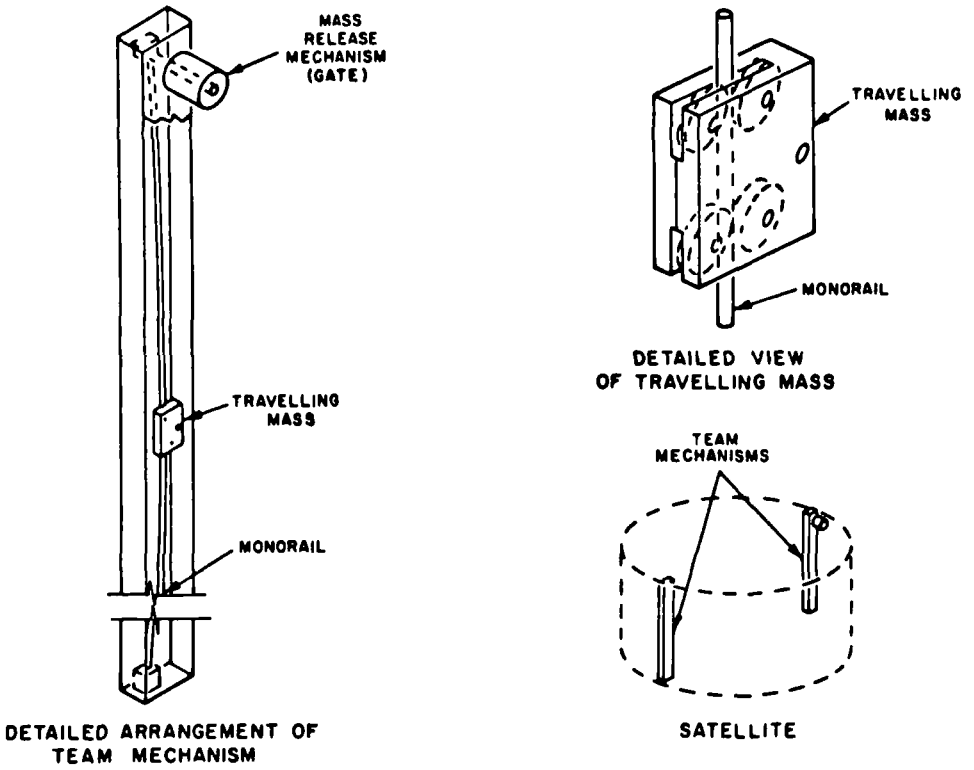


Figure II-32. TEAM Precession - Damping Devices

3. Despin Device

When the satellite separates from the third stage of the carrier rocket, its spin rate is approximately 125 rpm. To reduce this spin rate to an operational level in as short a time as possible, a despin mechanism is used to transfer the necessary amount of kinetic energy away from the vehicle. This despin mechanism consists of a pair of cable-attached masses that are released from the satellite structure after absorbing the necessary energy. The attachment points to the cables are located 180 degrees apart, and the cables wrap around the satellite a full 360 degrees, at a position approximately 8 inches above the baseplate. The masses are released at the required time by automatic firing of squibs.

Manual firing of squibs from the ground station, utilizing switch contacts on the rocket-firing switch in the satellite, is also provided as a back-up control measure. Centrifugal force causes each cable to unwind until it extends radially from the struture, at which time it slips off an open hook and departs from the satellite. This device reduces the spin rate of the satellite from a nominal 125 rpm to within the 8- to 12-rpm range in approximately 0.5 second.

4 Spin-Up Rockets and Rocket-Firing Switch

Ten spin-up rockets are mounted in diametrically opposite pairs under the periphery of the baseplate to provide for ground-controlled "coarse" spin-up of the spacecraft. The rockets are oriented at 54 degrees with respect to the tangent of the baseplate periphery in order to ensure that the effective thrust will be at the level required for use in conjunction with the MASC coil, i.e., the magnetic spin-control device.

Each pair of rockets is connected through contacts of a stepping switch to an ignition-control circuit, which fires individual sets of the diametrically opposite pairs of rockets upon ground-initiated command.

The spin-up rockets are not essential to the operation of the satellite because the spin rate can be controlled by means of the MASC coil. However, the thrust (1.42 lb-sec ± 10% per rocket) available with the use of the spin-up rockets facilitates a more rapid initial spin-rate acquisition than is possible with the use of the MASC coil only.

The rocket firing switch is a three-deck, 12-position, commutating switch with an electrical component board attached to the switch housing.

The first two positions of the switch are assigned to the firing of the precession damper and the de-spin mechanism squibs in the event of a separation switch failure.

The next ten positions of decks 2 and 3 are used for rocket firing, one position to a rocket pair, with alternate switch positions not used. The contacts on deck 2 place one side of each rocket pair in series with the 20-volt, 2-ampere rocket-firing power source from the auxiliary control unit. Deck 3 differs from deck 2 in that 11 of the 12 contacts are connected together and wired to ground, with one contact opened by a notch in the rotating contact of the wafer. This notch selectively lifts the shorting position from each rocket, allowing the 20-volt, 2-ampere rocket fire command to ignite the rocket. The gap width of the notch is such that two adjacent contacts will "break before make", preventing the possible shorting of the rocket-fire command if the switch does not step fully.

Deck 1 of the switch is assigned to telemetry. At each of the positions of the switch, a point on a voltage divider (consisting of eleven series-connected 1-Kohm resistors) is connected to the telemetry system, providing position indication. As the switch is stepped, the telemetry voltage will vary from 0 to 5 volts, in increments of 0.45 volts.

5. Dycon Units

a. General

Two redundant Dycon units are associated with the ESSA 1 dynamics-control subsystem, providing the following basic functions:

- programmed control of the QOMAC coil,
- programmed control of the MASC coil, and
- back-up camera vertical drive and shutter control.

Each Dycon can receive data from either of two redundant horizon-scanner pairs, as selected by ground command; and the horizon pulses are used by the Dycon for spin-synchronization of Dycon operation.

The Dycon receives horizon information from a horizon scanner, through a threshold amplifier. The optical axes of the horizon scanners and cameras are so oriented that each camera is pointing at the local vertical when the associated scanners are "seeing" a sky-to-earth transition. This orientation includes compensation for

system time delays, such as mechanical shutter delay. A horizon-scanner/threshold-amplifier combination 180° out-ot-phase (optically) with the camera-associated combination is used to generate one-half rotation information.

The Dycon is commanded and programmed from the ground stations by means of the command programmer. An OFF command turns off all power to the Dycon except "standby" power. For QOMAC- and MASC-coil operation, an ON command is first sent to the Dycon unit to condition the unit to receive programmed data. A command program, consisting of 18 data bits, is then sent to the selected Dycon.

The Dycon provides the programmed direct current for the QOMAC torquing coil. The basic attitude-correction cycle comprises the application of direct current through alternate halves of the center-tapped torquing coil, respectively, for two consecutive one-quarter orbit periods. Quarter-orbit measurements are generated by the Dycon on the basis of a pre-set orbit time, which is corrected after the satellite is in orbit.

The Dycon also provides the programmed direct current for the MASC coil. A complete spin-correction cycle comprises the flow of current through the coil for a quarter-orbit period and the removal of the current during the next quarter-orbit period. During the first quarter-orbit, direct current is applied through alternate halves of the center-tapped coil, respectively, for each half of a satellite rotation. Quarter-orbit measurements are generated by the Dycon on the basis of the pre-launch predicted orbit period. The one-half rotation information is generated by opposing horizon scanners, and the current through the torquing coil is phased to provide either spin-up or spin-down, with the appropriate change in spin rate determined by which end of the coil is first connected to ground.

As noted previously, the Dycon is also utilized to provide vertical-drive pulses for the camera and timing pulses for shutter operation, camera readout, and tape-recorder operation. The vertical-drive pulses are synchronized with the satellite spin-rate.

Synchronization of the vertical-drive pulses with the spin period (which is held constant at an integral multiple of 2.048-second vertical-scan times) permits a picture to be taken within approximately one satellite spin following a picture request. The shutter-triggering pulse is synchronized with a horizon-scanner signal to ensure that a picture is taken when the camera's line of sight is at the midpoint of its earth viewing time.

Further detail on the Dycon operation can be found in the TIROS IX Final Engineering Report.

b. Equipment Description

A Dycon unit is basically comprised of (1) a 17-stage shift/count register, (2) a state sequencer (4 control flip-flops), and (3) miscellaneous logic configurations and interface circuits. Figure II-33 is a block diagram of a Dycon unit. When an ON command is received, the register is enabled for data shifting. Upon receipt of the 17th data bit following a data "1", the register overflows and is enabled for counting. When the Dycon has been commanded in an attitude-control or a spin-control mode, T_O is counted out at 0.5 pps, in the least significant 11 stages. A maximum T_O of 1 hour, 8 minutes and 16 seconds can be counted out, in 2-second increments. At the end of T_O, attitude or spin-correction currents flow in the appropriate coil, and the register continues counting at 0.5 pps input rate. A feedback gate, whose inputs are connected by means of an externally mounted plug (one-quarter orbit plug) wired in conformance with the predicted orbit time, constrains the least significant 11 stages to count the time required for one quarter orbit, as determined by the one-quarter orbit plug. The one-quarter orbit time is adjustable in 4-second increments, with a minimum time of 17 minutes, 8 seconds and a maximum time of 34 minutes, 8 seconds. The 12th, 13th, and 14th stages of the register tally one-half orbit counts (torquing cycles). An overflow from these stages ends a torquing sequence, and all but standby power to the unit is turned off.

During an attitude-control sequence, after T_O the state of the 11th flip-flop denotes odd or even quarter orbits and controls which half of the center-tapped QOMAC coil receives current.

During a spin-control sequence, after T_O, the state of the 11th flip-flop determines whether or not the MASC coil receives current. The 15th flip-flop, whose state is programmed, and a control flip-flop, which receives information from opposite-looking horizon sensors, determine the phasing of currents in each half of the MASC coil.

When the Dycon is in the camera-triggering mode, the least significant 10 stages of the register count at a 500-pps input rate, allowing vertical drive pulses to be initiated at 2.048-second intervals. The register is decoded to generate a vertical retrace time of 128 milliseconds, within the 2.048-second interval.

Timing for the Run Tape signal is also decoded from the register. Register stages 14, 16, and 13, respectively, store and actuate the picture request, and control the 10-kc modulator.

The following inputs are supplied to the Dycon units:

(1) "power on" and "power off" commands (from the -13 volt regulators),

(2) data "1"s and "0"s,

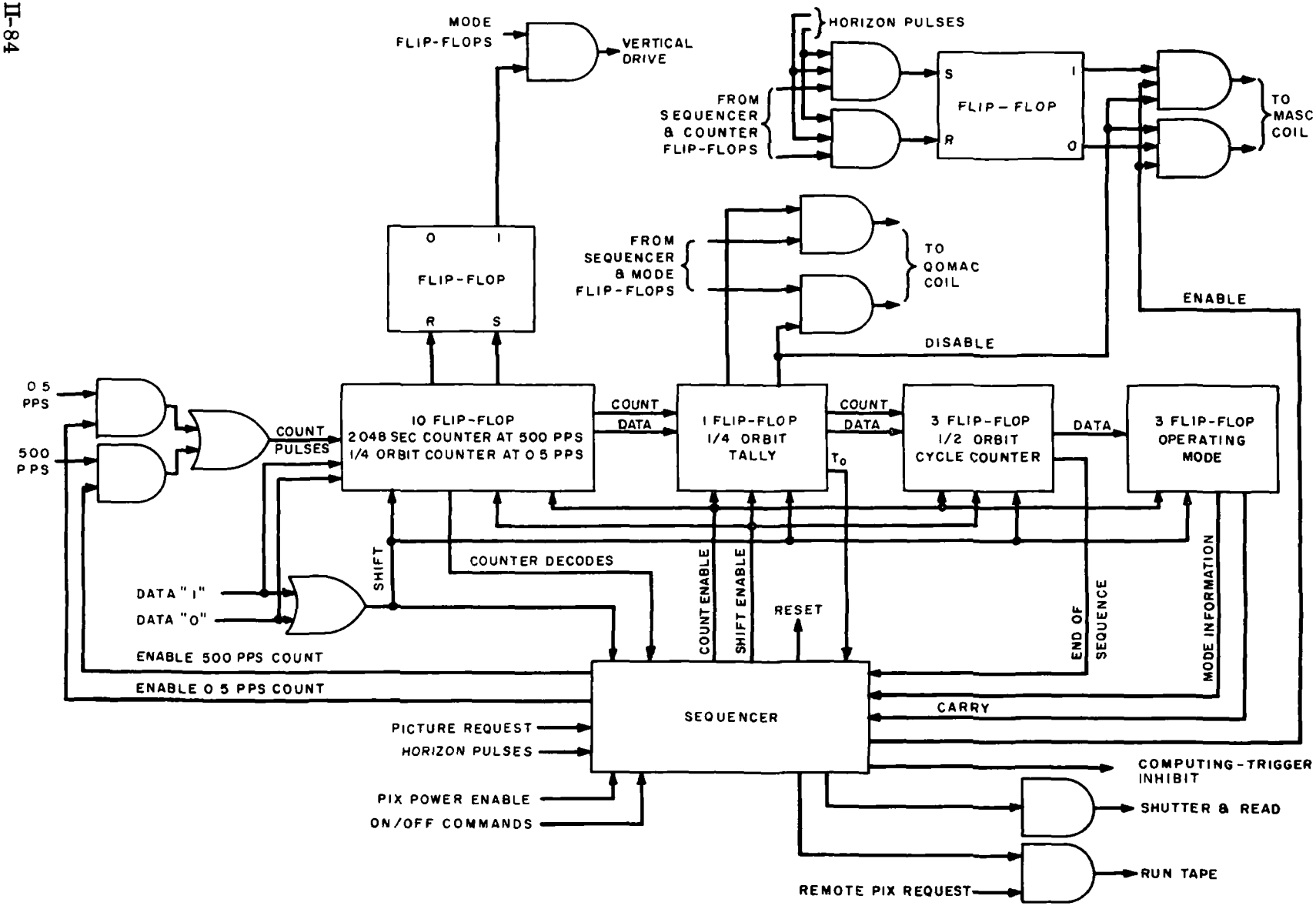


Figure II-33. Dycon Unit, Block Diagram

(3) 0.5-pps register counting signal in attitude and spin control modes,

(4) 550-pps register counting signal in camera-triggering mode,

(5) picture regulated power (from the -24.5 volt regulators),

(6) remote picture requests,

(7) horizon pulses from both scanners in the selected horizon-scanner pairs, and

(8) run-tape inhibit signal.

The following outputs are supplied by the Dycon Units:

(1) vertical drive pulses for the selected camera,

(2) shutter and read signals for the selected camera,

(3) Run-Tape signal,

(4) 10-kc modulator signal,

(5) data-verification signal,

(6) computing-trigger inhibit signal,

(7) QOMAC-coil operating currents (to one-half of the coil),

(8) MASC-coil operating currents (to one-half of the coil), and

(9) telemetry outputs.

c. Dycon Command Programs

The function to be performed by the Dycon is selected by ground command. In the case of attitude-control and spin-control functions, the torquing-current flow program is also determined by ground command. The camera-triggering function can be performed by either the computing trigger or the Dycon. If the Dycon is performing this function, the computing trigger is disabled by an inhibit signal from the Dycon. Once in the triggering mode, the Dycon need not be programmed for successive picture-taking operations, because a command is required to change it from this mode.

A programmed Dycon command is preceded by data bits which perform the following functions:

- carry the sync signal (four data bits) required to enable decoder operation in the camera clock so that the message address bits, which follow the sync signal, can be decoded, and
- the message address bits (five data bits) which enable the selected Dycon unit.

The actual command for the selected Dycon unit consists of 18 data bits, the least significant 17 of which comprises the actual data required. The most significant bit is always a "1"; the overflow of this bit indicates that the programming is completed. The possible command programs for the Dycon are given in Table II-5.

Each Dycon unit can be used for only one function at a time. However, since ESSA 1 includes two, redundant Dycons, along with two, redundant computing triggers, the satellite is capable of performing any of the following combinations of functions at one time:

- attitude control or spin control, and computed triggering of both cameras;
- attitude control, computed triggering, and Dycon triggering; spin control, computed triggering, and Dycon triggering; and Dycon triggering of both cameras.

6. QOMAC Coil

a. General

The QOMAC coil consists of aluminum wire wound about a fiberglass frame and bonded to the frame. The frame is secured by braces to the top of the inside of the hat. The QOMAC and MBC coils are wound on the same frame, and the complete unit weight 6.62 pounds.

The basic QOMAC-coil torquing cycle consists of the application of direct current through alternate halves of the center-tapped coil for two consecutive quarter-orbit periods. The command word sent to the Dycon specifies which Dycon is to be used and which Dycon mode. If QOMAC-or MASC-coil operation is specified, the command also specifies when the torquing cycles should begin, and how many cycles should be performed.

b. Functional Description

When the counting operation begins, the programmed delay time is counted out at a rate of 0.5 pulse per second. At the end of the delay time, that half of the QOMAC coil is energized which will produce a positive magnetic dipole when connected to ground. At the same time, the register beings counting quarter-orbit periods.

The quarter-orbit period is pre-wired in the unit based on an orbit time of 100 minutes. When 25 minutes (approximately equal to the predicted quarter-orbit period) has been counted, the polarity of the QOMAC dipole is reversed; the ground connection is switched to the other half of the coil and is permitted to remain there for another quarter-orbit period.

A QOMAC cycle can be defined, therefore, as a positive dipole for 25 minutes followed by a negative dipole for 25 minutes. The number of switchings is counted and, after the programmed number of QOMAC cycles has taken place, the coil current is turned off automatically and the Dycon unit returns to a standby condition.

7. MBC Coil

a General

The MBC coil is wound about the same frame as the QOMAC coil. As noted, this frame is secured to the top of the inside of the spacecraft hat, and the frame is mounted orthogonal to the MASC coil.

The MBC coil is used to cancel the inherent magnetic dipole within the satellite caused by interaction of the earth's magnetic field and the ferromagnetic materials used in the satellite's construction.

b. Functional Description

After a number of days in orbit, the satellite begins to show an attitude drift due to interaction of the residual dipole with the earth's magnetic field. This drift is analyzed, the residual dipole is calculated, and the MBC-coil switch is commanded by the ground station to step to a position to cancel the residual dipole. The MBC coil is then further used to generate a constant dipole to cause the spacecraft attitude to drift sufficiently to compensate for the precession rate of approximately 1 degree per day that occurs with a sun-synchronous orbit. This constant, controlled drift reduces the amount of QOMAC programming required.

The MBC-coil switch is a solenoid-operated, three-deck, twelve-position stepping switch that is used to control the amount of current in the MBC coil, and, thereby, the

magnitude of its magnitude dipole. At each of 10 positions on deck 1, a current-limiting resistor is placed in series with the MBC coil and the 13-volt d-c power supply, enabling the selection of magnetic moments from 0.2 to 2.0 ampere-turns-meter^2, in increments of 0.2 ampere-turns-meter^2.

Deck 2 of the switch is assigned to the telemetry function; and, at each position of the switch, a point on a voltage divider is connected to the telemetry system in order to provide an indication of switch position to the ground stations. The telemetry voltage varies from 0 to -5.0 volts, in increments of 0.45 volt.

Deck 3 is assigned to functions involved with over-all dynamics control of the satellite. Positions 1 and 2 on this deck are for Dycon selection; position 11 is for the high-torque mode of QOMAC-coil operation; position 12 is the "home" position. When the switch is in either position 11 or 12, no current will flow in the MBC coil.

8 MASC Coil

a. General

The MASC coil is comprised of aluminum wire wound in a fiberglass frame and bonded to the frame with epoxy. The weight of the unit is 2.76 pounds. The MASC coil is supported inside the spacecraft hat by means of struts and braces; however, the braces are not bonded to the frame, so as to produce, thereby, a relatively "soft" suspension of the coil.

The MASC coil is operated by means of the Dycon units, which, through the MASC switch, provide programmed switching of the center-tapped coil, controlling both the direction of magnetic dipole induced by the coil and the period of current flow. The MASC switch also allows for read-out of switch position.

b. Functional Description

The general operation of the MASC coil is similar to the operation of the QOMAC coil. In addition to quarter-orbit switching periods, half-spin switching periods synchronized with sensing of the Earth's horizons are also utilized. Furthermore, the quarter-orbit switching does not result in a polarity reversal. Instead, the initial quarter-orbit polarity is established in response to the command message, and the current to the entire coil is turned on and off at the quarter-orbit intervals. During the quarter-orbit period that the coil is on, the polarity of the magnetic dipole is reversed at half-spin intervals by alternately switching the ground connection to opposite ends of the center-tapped coil, with the choice of which half of the coil is first connected to ground determining whether an increase or decrease in spin rate is produced.

A MASC cycle is, therefore, defined as an alternate switching cycle for 25 minutes, followed by an OFF period of 25 minutes. The half-orbit cycles are repeated as many times as programmed, after which the coil current is turned off and the Dycon unit is automatically switched to the standby condition.

9. Testing

a. Lift-Off and Separation Switch, MBC Switch, and Rocket Switch

The lift-off and separation switch, Serial No. 3-524-22, was subjected to vibration testing on September 5, 1963. The MBC switch, Serial No. 3-527-03, was vibrated on May 7, 1964, while the rocket switch, Serial No. 3-525-03, was vibrated on May 13. In each case, the switch passed vibration testing with satisfactory results; and all three switches were subjected to thermal-vacuum testing with the integrated spacecraft.

b. Precession Dampers

In the course of stiction, i.e., "breaking" friction, tests performed in May and June, 1964, it was found that the stiction was excessive on both precession dampers, and the results of subsequent tests indicated excessive frictional drags on the damper carts. This condition was attributed to flat spots or "Brinelling" of the bearings used in the damper carts, and all bearings were replaced by selected bearings. The stiction value was again measured in July, and found to be 114 miligrams for the Serial No. 3-523-10 damper and 102 miligrams for the Serial No. 3-523-04 damper. A stiction force of 200 milligrams results in a nutation damping threshold of 0.5 degree; therefore the dampers were considered capable of reducing satellite nutation to well below the required value of 0.5 degree.

Precession damper No. 1, Serial No. 3-523-04, was successfully subjected to vibration testing on August 4, 1964; damper No. 2, Serial No. 3-523-10, passed vibration testing on August 5, 1964. The stiction forces were measured both before and after vibration of each damper and the values were found to be within requirements.

Spacecraft OT-3 had been first fitted with Serial Nos. 3-523-13 and -16 dampers; however, on December 8, 1964, during spin testing of the spacecraft, all squibs were accidentally fired. The dampers were then replaced, along with the squibs, with Serial Nos. 3-523-04 and -10.

On December 5, 1965, during a SEPET test on the integrated spacecraft, the squibs were again accidentally fired. On this occasion, however, the hat was inclined, preventing the damper cart from moving from its normally stored position. Consequently, it was only necessary to replace the squibs and reinsert the retainer pins into the carts.

c. Despin Mechanism

The despin mechanisms, Serial Nos. 3-526-07 and 08, were successfully subjected to vibration testing in November 1963; thermal-vacuum testing of the despin mechanisms was achieved with testing of the integrated spacecraft.

d. QOMAC, MBC, and MASC Coils

The magnetic-torquing coils were subjected to environmental testing with the integrated spacecraft with satisfactory results.

e. Dycons

The ESSA 1 dycon units, Serial Nos. 3-623-02 and -03, were initially subjected to vibration testing on May 4, 1964, and thermal-vacuum testing on May 7, with satisfactory results. However, these units were modified along with the TIROS IX dycons because of conditions* which arose during the TIROS IX test program. Because of the modifications it was decided to re-vibrate the dycons (in the thrust axis only). This additional vibration was successfully performed on both units on July 14, 1964.

I. SATELLITE STRUCTURE

1. Component Layout

The structure used for spacecraft OT-3 was identical to that used for TIROS IX and basically similar to the structures used for all other TIROS satellites. The structure comprises a baseplate assembly, upon which operating components are mounted, and a cover assembly, i.e., the "hat", which is used as a mounting surface for the solar-cell array. The structural differences between the wheel satellites, ESSA 1 and TIROS IX, and the standard TIROS satellites related to the use of radial camera assemblies, rather than axial, for the wheel satellites.

The cameras in a wheel satellite view the earth through diametric sides of the hat, rather than through the baseplate. Each camera is mounted as far outboard on the baseplate as practical to minimize the loss of solar-cell mounting area and the consequent loss of power.

*For a detailed description of the problem encountered and the resultant modification, see the TIROS IX Final Engineering Report.

The design weight for the satellite was projected at a nominal value of 300 pounds (balanced). The weights of the spacecraft OT-3 components are listed in Table II-6 along with the respective Serial NOs. of the units and their angular locations on the baseplate. Figure II-34 shows the layout of the spacecraft OT-3 components, which was dictated by the electrical system requirements and by the requirements of dynamic balance.

The two cameras were placed 180 degrees apart, with the two tape transports also 180 degrees apart and the center lines of the tape transports perpendicular to those of the cameras. The batteries were located at the center of the baseplate, along the spin axis, and the remaining components (which were, for the most part, redundant units) were arranged symetrically about the spin axis, with due consideration given to establishing a reasonably large ratio either of I_s/I_{yaw} or I_s/I_{pitch}.

The earth intercept (126 degrees at a 400 nautical mile circular orbit) and system considerations inherent in the use of the computing triggers and the camera-triggering principle imposed an additional consideration on the placement of the two sets of horizon scanners.

Centerlines were accurately scribed onto the camera wedges. These wedges were then installed onto the spacecraft so that the scribe lines were in phase. These wedges were optically checked, and then shimmed until the plane of each was at the calculated value in relation to the separation plane. The cameras were then installed and the alignment adjustment was made between each camera and its wedge.

The horizon scanners were then installed on structural beams mechanically set at a pre-determined angle with relation to the cameras. The mechanical and optical axes of the scanners were then shimmed to the structural beams until the desired optical relationships to the cameras were obtained. The V-head attitude sensor was shimmed to the structure until the separation plane was at the mid-point between the optical axes of the two "heads."

Circular boxes were designed for the spaces beneath the tape recorders and this space was used for the power distribution units. Damping bracketry was added to all of the taller boxes; this bracketry consisted solely of aluminum sheets (0.05-inch thick) bolted to the boxes.

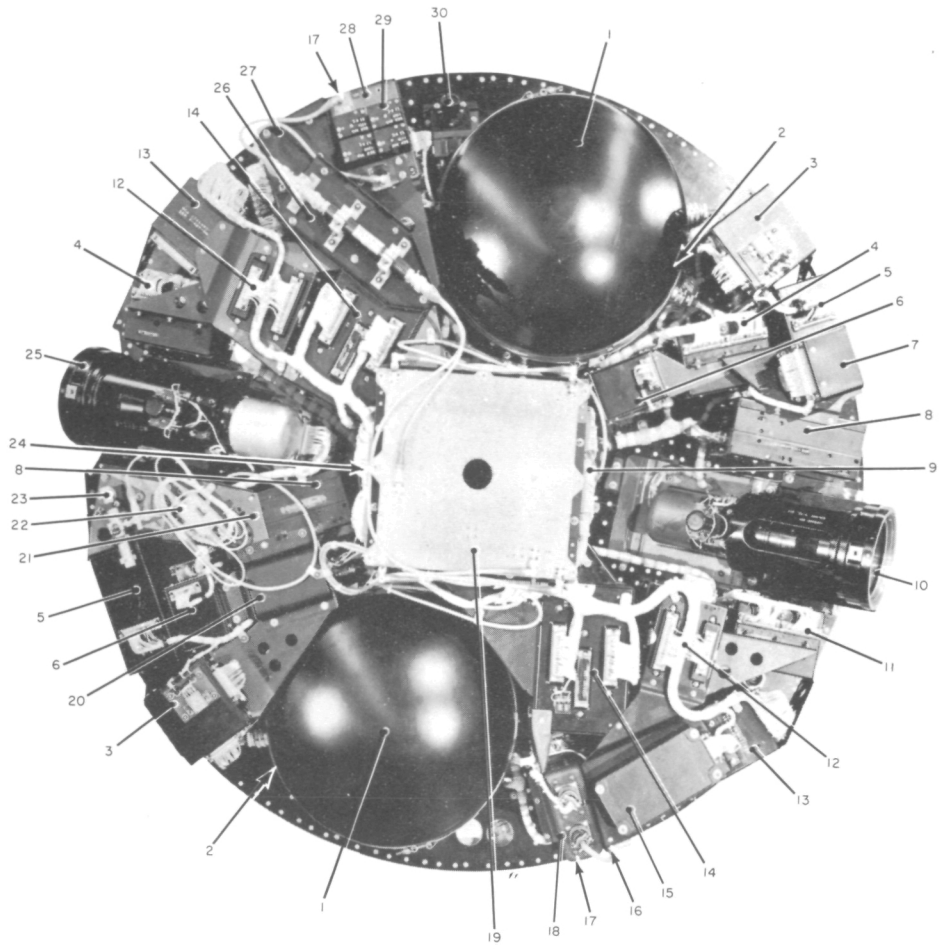
As noted throughout this report, the layout and design of spacecraft OT-3 were almost identical to those of TIROS IX, the first wheel satellite. Table II-7 presents a comparison of the final mechanical measurements for each.

TABLE II-6. SERIAL NUMBERS, ANGULAR LOCATIONS, AND WEIGHTS OF SPACECRAFT OT-3 COMPONENTS

Component	Serial No.	Weight (GMS)	Radial Location
Antenna Coupler	3-424-04	937	Center
Antenna Filter Assembly	03	42	150°
Antenna, Receiving	3-425-08	119	Center
Antenna, Transmitting	3-426-04	896	Center
Attenuator	AA-031T	32	220°
Attenuator	AA-03T	32	220°
Auxiliary Control Unit	3-624-5	1098	5°
Bandpass Filters	3-423-02,03	99/99	220°
Battery Pack	3-721-08	17924	Center
Beacon Transmitters/SCO's Assembly	3-423-03	1257	235°
Camera No. 2	3-221-11	3295	350°
Camera No. 1	3-221-08	3253	170°
Camera Control No. 1	3-621-03	977	180°
Camera Control No. 2	3-622-02	996	320°
Camera Electronics Unit	3-222-8,11	1850/1803	170°, 335°
Command Address Unit	3-626-3	465	40°
Command Distribution Unit	3-627-01	1390	Center
Command Receiver Unit	3-422-01	1110	155°
Camera Clocks Nos. 1, and 2	3-625-4,5	3581/3462	40°
Commutators/Signal Conditioner	3-925-03 (Commutator Serial Nos. 229, 305)	2740	220° 220°
Computing Triggers	3-629-4,5	1145/1145	200°, 20°
De-Spin Mechanisms	3-526-7,8	686/687	90°, 270°
Dycon Units	3-623-02,03	1709/1698	200°, 150°
Fuse Block	3-728-03	43	40°

TABLE II-6. SERIAL NUMBERS, ANGULAR LOCATIONS, AND WEIGHTS OF SPACECRAFT OT-3 COMPONENTS (Continued)

Component	Serial No.	Weight (GMS)	Radial Location
Horizon Scanners (1A, 1B)	3-921-109, 112	212/215	232°
Horizon Scanners (2A, 2B)	3-901-113, 108	213/211	52°
Lift-Off Switch	3-524-22	447	Center
MASC Coil	3-521-02	755	-
MBC Switch	3-527-3	409	325°
QOMAC and MBC Coils (wound on same frame)	3-522-02	3091	-
Power Distribution Unit	3-726-2,4	219/211	110°
Power Supply Protection Unit	3-723-7	1049	40°
Precession Dampers	3-523-4,10	549/545	288°, 108°
Rocket Switch	3-525-03	326	170°
Solar Aspect Indicator Electronics Unit	3-923-03	640	55°
Solar Aspect Indicator Sensor	3-924-03	70	50°
Tape Recorder Power Convertors Nos. 1 and 2 (R-2 Boxes)	3-322-8,11	1439/1439	320°, 140°
Tape Recorder Signal Electronics Nos. 1 and 2 (R-3 Boxes)	3-323-10,11	1075/1092	120°, 300°
Tape Transports Nos. 1 and 2	3-321-13,12	4733/4730	270°, 90°
Threshold Amplifier	3-630-3,2	152/153	
TV Transmitters Nos. 1 and 2	3-421-18,4	791/777	130°
V-Head Attitude Sensor	3-922-104	433	246°
Video Coupler	3-427-02	57	130°
Voltage Regulators Nos. 1 and 2	3-727-4,5	815/809	320°



LEGEND

1. Tape Recorder
2. Power Control Unit (Hidden)
3. Recorder Electronics
4. Command Control Unit
5. Recorder Power Converter
6. Voltage Regulator
7. MBC Switch
8. Camera Electronics
9. Command Distribution Unit
10. TV Camera No. 2 (Viewing 26.5° Upward)
11. Auxiliary Control Unit
12. Dycon Unit
13. Computing Trigger
14. Camera Clock
15. Power Supply Protection Unit (Hidden)
16. Solar Aspect Indicator
17. Two Horizon Scanners (Hidden)
18. Solar Aspect Indicator Electronics
19. Antenna Coupling Network
20. TV Transmitters and Video Coupling Network
21. Command Receivers
22. 4-Port Hybrid for Command Receivers
23. Notch Filter
24. Battery Pack (Hidden)
25. TV Camera No. 1 (Viewing 26.5° Downward)
26. Filters and Attenuators for Beacon Transmitters
27. Telemetry Commutators and Signal Conditioner
28. Beacon Transmitters
29. Subcarrier Oscillators
30. V-Head Attitude Sensor

Figure II-34. Component Layout, Spacecraft OT-3

TABLE II-7. COMPARISON OF MECHANICAL MEASUREMENTS OF SPACECRAFT OT-3 AND TIROS IX

Mechanical Measurement	Spacecraft OT-3	TIROS IX
Final Weight*	301.56 lbs.	300.45 lbs.
Moments of Inertia		
• About Spin Axis*	164.028 lb-in-sec^2	164.55 lb-in-sec^2
• About Spin Axis**	166.740 lb-in-sec^2	167.07 lb-in-sec^2
Maximum Transverse		
Moment of Inertia**	117.926 lb-in-sec^2	112.05 lb-in-sec^2
Center of Gravity*		
(above separation plane)	10.44 in	10.40 in

*Space configuration: de-spin weights not included.
**Including de-spin weights.

2. Thermal-Design Analysis

Since spacecraft OT-3 was almost identical to TIROS IX, it was not required that a new thermal-design analysis be performed. The analysis performed on the TIROS IX program showed that the thermal design of the wheel spacecraft was satisfactory for the spacecraft OT-3 mission and that all spacecraft components would be maintained within the desired temperature limits.

The thermal-design analysis performed for the TIROS IX program consisted of (1) a 3-body computer analysis, (2) an n-body computer analysis, and (3) a thermal-vacuum test of the thermal model. In addition, a separate analysis was conducted of the predicted component temperatures during the satellite torquing phase and a series of emissivity tests was performed on the TIROS IX baseplate.

Further detail on these studies can be obtained from the TIROS-wheel design-study report* and from the TIROS IX Final Engineering Report.

*Design-Study Report on the TIROS-Wheel Spacecraft with Canted TIROS Cameras, Astro-Electronics Division, Radio Corporation of America, R-2223, 15 January 1964.

3. Temperature Sensors

Fourteen temperature sensors were mounted on the spacecraft to permit an accurate means of monitoring actual component operating temperatures while the spacecraft is in orbit. The locations of the sensors on spacecraft OT-3 and the operating range of each are presented in Table II-8. As on TIROS IX, temperature sensors measuring either of two ranges, -10 to +50° C or -30 to +100° C, were utilized.

4. Mechanical Integration

The assembly procedures involved in the construction of spacecraft OT-3 were the standard procedures utilized on all previous TIROS programs. Upon receipt of the structure from the manufacturer, the mechanical integration activity was initiated. The baseplate was first prepared to receive the subsystems. Radial lines were laid out on the baseplate to facilitate alignment of the units. Mounting holes and grounding holes were then drilled for each box, and the bracketry in the hat was prepared.

Some subsystems were mounted on the baseplate before the harness was installed. When the harness fit was determined to be satisfactory, most of the remaining boxes were installed. Some boxes were left off the baseplate at that point to afford access for electrical integration purposes. As electrical integration proceeded, the remaining boxes were added.

After the completion of electrical integration, the boxes were temporarily tied down to permit electrical debugging to be performed. With the successful completion of electrical debugging, the final tie-down was performed, the inter-box bracketry was added, and the coaxial cabling was installed.

Table II-9 presents a chronological listing of the major events in the mechanical integration of spacecraft OT-3.

TABLE II-8. LOCATION OF TEMPERATURE SENSORS ON SPACECRAFT OT-3

Sensor No.	Location	Radius	Radial	Operating Range
1	TV Transmitter	-	-	narrow*
2	QOMAC Coil	-	-	narrow
3	Solar Cell (Top Hat)	3 in	250°	wide**
4	Inner Top Skin	3 in	70°	wide
5	Inner Top Skin	12 in	10°	wide
6	Side Panel	6 in from top of hat	10°	wide
7	Side Panel	6 in from bottom of hat	10°	wide
8	Top of Baseplate	14 in	30°	narrow
9	Battery Pack	-	-	narrow
10	Camera No. 1	-	170°	narrow
11	Camera No. 2	-	350°	narrow
12	Digital Clock	-	50°	narrow
13	Beacon Transmitter	-	-	narrow
14	Voltage Regulator	-	310°	narrow

*-10 to +50°C
**-30 to +100°C

TABLE II-9. MAJOR EVENTS IN THE MECHANICAL INTEGRATION OF SPACECRAFT OT-3

Date	Activity
4/27/64	Work started on preparing baseplate and hat for integration of spacecraft subsystems.
8/8/64	QOMAC and MBC coils fitted in hat.
8/25/64	All mounting holes in baseplate drilled and cleaned; temporary mounting of boxes initiated.
8/27/64	Harness fitted and installed.
8/31/64	Harness installed under baseplate.
9/14/64	Installation of inter-box bracketing initiated.
10/6/64	Spacecraft prepared for electrical integration.
10/7/64	Spacecraft turned over to electrical integration.
10/13/64	Temperature sensors installed on spacecraft.
11/18/64	Final tie-down of boxes and addition of inter-box bracketry initiated.
11/24/64	Basic mechanical integration completed, and RF cables installed; at this point the spacecraft was ready for the initial alignment checks and initial measurements in the qualification test program.
2/12/65	Spacecraft placed in storage.
2/65 to 10/65	Spacecraft checked on a bi-weekly basis.
10/18/65	Spacecraft removed from storage
11/18/65	Spacecraft mounted on model 3rd-stage launch vehicle for RF-interference tests and command sensitivity tests.
11/22/65	Flight-model battery pack installed.

TABLE II-9. MAJOR EVENTS IN THE MECHANICAL INTEGRATION OF SPACECRAFT OT-3 (Continued)

Date	Activity
12/2/65	Spacecraft taken outdoors for solar-cell array test.
12/3/65	One-year timing devices removed from spacecraft and replaced with weights.
1/4/66	4-Port hybrid matching circuit added to spacecraft and notch filter relocated.
1/7/66	Despin weights and cables added.
1/14/66	Spin-up rockets and new despin weights installed; spacecraft placed in shipping container for delivery to launch site.

SECTION II. SPACECRAFT TESTS

A INTRODUCTION

Each unit on spacecraft OT-3 was extensively tested to ensure that it met the requirements of the TIROS subsystem prototype- and flight-level specifications before being accepted for integration with the spacecraft. After all units had been integrated on the spacecraft and electrical checkout had been completed, the test program on spacecraft OT-3 was initiated. By NASA directive, the test program was scheduled in two phases, with Phase A to be performed before the spacecraft was placed in a "hold" condition and Phase B when the "hold" was rescinded.

Phase A included the initial procedures for the alignment of the spacecraft sensors, the balancing of the spacecraft, and the dynamic spin test; it also included sine and random vibration tests, and thermal-vacuum tests at 0° and 50° C. Phase B included the final alignment, balancing, and dynamic spin-test procedures, random vibration tests, thermal-vacuum tests at 0°, 15°, and 35°C, and final checks and measurements.

The Environmental Test Committee, comprising four engineering representatives from NASA and three from AED, was established before the start of the environmental test program, to review and approve spacecraft performance and test results. The first phase of mechanical integration of spacecraft OT-3 was completed on October 6, 1964, and electrical debugging was then performed. Initial checks and measurements were initiated on December 2, 1964, and completed on December 8.

The Phase A environmental qualification tests were initiated on December 9, 1964. These tests were completed on January 15, 1965; and, from February 12, 1965, to October 18, 1965, the spacecraft was maintained in a "hold" status by NASA directive. The Phase B tests were initiated on December 11, 1965, and completed on December 31. On January 14, 1966, with the conclusion of final checks and measurements, NASA declared spacecraft OT-3 ready for delivery. On January 17, 1966, the spacecraft was shipped to the launch site at the Eastern Test Range.

After its arrival at the launch site, the spacecraft was tested on a regular basis. Based on the performance of the spacecraft during these tests, NASA accepted the spacecraft as being flight-qualified. Spacecraft OT-3 was successfully launched and orbited on February 3, 1966.

B. COMPONENT TESTS

In line with the test philosophy used for the TIROS programs, all new components were subjected to prototype-level testing. Upon receipt of each group of purchased satellite parts, a complete quality inspection and test was conducted by the Purchase Material Inspection activity to ensure that the parts met specification requirements.

It was required that the prototype model of each new component satisfactorily complete the following series of tests:

(1) Vibration

  (a) Thrust Axis: 21 g rms, 2 minutes, 20 to 2000 cps.

  (b) Lateral Axis No. 1: 15 g rms, 2 minutes, 20 to 2000 cps, and

  (c) Lateral Axis No. 2: 15 g rms, 2 minutes, 20 to 2000 cps.

(2) Sustained Acceleration (Positive and Negative)

  (a) Thrust Axis: 50 g, 5 minutes,

  (b) Lateral Axis 1: 50 g, 5 minutes, and

  (c) Lateral Axis 2: 50 g, 5 minutes.

(3) Thermal Vacuum (5×10^{-5} mm Hg)

  (a) Hot cycle: +60°C, 12 hours,

  (b) Ambient cycle: +25°C, 12 hours, and

  (c) Cold cycle· -15°C, 12 hours.

The flight models of new components were subjected to vibrations of 10 g rms in the thrust direction and in each of two, mutually perpendicular lateral (transverse) axes. The frequency spectrum for these vibrations was 20 to 2000 cps, and the duration of each test was 2 minutes.

After vibration, the new or modified flight-model components were mounted in a thermal-vacuum chamber (5×10^{-5} mm Hg) and checked for satisfactory operation at the following temperatures, for the periods indicated·

- -10°C for 12 hours,
- +25°C for 12 hours, and
- +55°C for 12 hours.

C. QUALIFICATION TESTING OF SPACECRAFT OT-3

1. General

The qualification test program on spacecraft OT-3 was preceded by a complete SEPET* (Standard Electrical Performance Evaluation Test) performed to provide a thorough electrical checkout of the spacecraft; a SEPET was then performed at the completion of each major phase of the test program.

The environmental qualification tests were preceded by the initial checks and measurements; at the completion of qualification testing these checks and measurements were repeated, and a mechanical inspection of the spacecraft was performed.

The environmental test requirements for vibration and thermal-vacuum testing of spacecraft OT-3 are listed in Table II-10. The sequence of activities followed in the test program was in accordance with qualification test procedures and was as follows·

(1) SEPET

(2) Initial Alignment

(3) Initial Balancing

(4) Initial Dynamic Spin Test

(5) Vibration Test, Part A

(6) SEPET

(7) Thermal-Vacuum Test, Part A

(8) SEPET

(9) Vibration Test, Part B

(10) SEPET

(11) Thermal-Vacuum Test, Part B

(12) SEPET

(13) Final Alignment

(14) SEPET

(15) Final Dynamic Spin Test

(16) Final Balancing

(17) Final Measurements

(18) SEPET

* The SEPET performed on spacecraft OT-3 is described in Volume II of this report.

TABLE II-10. SPACECRAFT OT-3 ENVIRONMENTAL TEST REQUIREMENTS

THERMAL-VACUUM TESTING			
	Duration	Temperature	Pressure
PART A	72 hours	50°C	not exceeding 5×10^{-5} mm Hg
	72 hours	0°C	
PART B	12 hours	35°C	
	24 hours	15°C	
	48 hours	0°C	

VIBRATION TESTING			
	Axis	Type of Vibration	Vibration Level
PART A	Thrust	Sine	10 to 550 cps at 2.5 G peak; 550 to 2000 cps at 14 G peak — 4 oct/min
	Lateral Axes (two orthogonal axes)	Sine	10 to 500 cps at 1.5 G peak; 500 to 2000 cps at 2.5 G peak — 4 oct/min
	Lateral Axes	Random	7.7 G rms, 20 to 2000 cps, 2-min duration
PART B	Thrust	Random	7.7 G rms, 20 to 2000 cps, 2-min duration
	One Lateral Axis (80/260°)	Random	7.7 G rms, 20 to 2000 cps, 2-min duration

The flight-model battery pack was not integrated on the spacecraft for the Phase A environmental tests in order to avoid subjecting it to long exposure to high temperature, i.e., 72 hours at 50°C. (Such temperatures are, of course, not experienced by the spacecraft in the space environment.) A systems test model battery pack was installed in its place for the Phase A tests; the flight-model battery pack was then installed on the spacecraft for the Phase B tests, after the "hold" had been rescinded.

Before the Phase B tests were, in fact, performed, they were revised as follows:

Test	As Originally Specified	As Revised
Thermal Vacuum	24 hours at 35°C	12 hours at 35°C
	48 hours at 0°C	24 hours at 15°C
		48 hours at 0°C
Vibration	Random Vibration in Thrust Axis Only	Random Vibration in both the Thrust Axis and one Lateral Axis

During vibration testing, power was applied only to those components that would be operative during the launch phase. During thermal-vacuum testing, the spacecraft was tested on an operational basis at 2-hour intervals, with the responses carefully monitored and recorded.

Table II-11 presents a summary of the various checks, measurements, and tests performed in the test program for spacecraft OT-3, and the respective completion dates.

2. Initial Checks and Measurements

a. Initial Alignment

The initial alignment procedures on the TV cameras, the V-head attitude sensor, the horizon scanners, and the solar-aspect indicator were all performed on December 2, 1964, with satisfactory results.

The procedures used in both the initial and final alignment of the units were almost identical and are detailed, along with the calibration procedures, in the Alignment and Calibration Handbook for spacecraft OT-3*.

*"Alignment and Calibration Data for the TIROS OT-3 Meteorological Satellite", Astro-Electronics Division, Radio Corporation of America, M-2080, 17 January 1966.

TABLE II-11. SUMMARY OF SPACECRAFT OT-3 TEST PROGRAM

Activity	Completion Date
Initial Alignment	12/2/64
Initial Spin Balancing	12/4/64
Initial Dynamic Spin Test	12/8/64
Vibration Test, Part A	12/10/64
Thermal-Vacuum Test, Part A	1/15/65
2/12/65 HOLD	10/18/65
Vibration Test, Part B	12/10/65
Thermal-Vacuum Test, Part B	12/31/65
Final Alignment	1/3/66
Final Spin Balancing	1/4/66
Vibration Test	1/7/66
Final Dynamic Spin Test	1/10/66
Final Spin Balancing*	1/11/66
Final Measurements**	1/12/66

* Performed because of modification to command receivers; see "History of Qualification Testing" later in this section.

** Including magnetic dipole moments; TV-camera brightness response and distortion and centering; and moments of inertia, center of gravity, and final spacecraft weight.

b. Initial Balancing

On December 4, 1964, initial balancing of spacecraft OT-3 was performed. The balance weights added to the spacecraft were as follows:

- Baseplate: 306 gms at the 329-degree radial.
- Hat (inside): 340 gms at the 250-degree radial.

- Hat (outside)· 50 gms at 282-degree radial;
  7 gms at 215-degree radial.

c. Initial Dynamic Spin Test

The initial dynamic spin test was performed on December 8, 1964, with satisfactory results. The purpose of this test was to verify proper operation of the components that are used to initiate and control spin-triggering of the TV cameras, and the effects of varying spin rate, attitude, and roll-angle error were tested. The procedures used in both the initial and final dynamic spin tests were identical; the tests were performed in the Dynamic Spin Facility, which was designed and built by AED and first used for the TIROS IX testing program.

3. Vibration Testing

Part A of the vibration phase of environmental qualification testing of spacecraft OT-3 was performed on December 10, 1964. The spacecraft was subjected to the types of vibration and the vibration levels specified in Table II-9, with satisfactory results.

On December 11, 1965, after the "hold" on spacecraft OT-3 had been rescinded, Part B of vibration testing (as specified in Table II-10) was successfully performed. This test completed the vibration requirements for environmental qualification of spacecraft OT-3. On January 7, 1966, however, it was necessary to perform additional vibration testing of the spacecraft because of modifications made to the spacecraft after environmental testing had been completed. (As detailed in the History of Qualification Testing, these modifications consisted of the addition of a hybrid matching circuit and a new RF cable.) The additional vibration performed at this time consisted of random vibration in the thrust axis (7.7 g rms, 20 to 2000 cps, 2-min duration). This test was performed with satisfactory results.

4 Thermal-Vacuum Testing

Part A of the thermal-vacuum phase of environmental qualification testing was initiated, at the 50°C test level, on December 12, 1964. Testing at this temperature was completed on January 13, 1965, after several interruptions due to various problems* which arose during checks of spacecraft operation. On the same day, testing at the 0°C test level was initiated. This test was completed on January 15, with satisfactory results, thus fulfilling the Part A requirements for thermal-vacuum testing.

*The problems encountered during thermal-vacuum testing and referenced herein are detailed, along with the associated solutions, in the History of Qualification Testing.

By NASA directive, the spacecraft was placed in a "hold" status on February 12, 1965, and maintained in that status until October 10, 1965. On December 14, 1965, after the "hold" had been rescinded, Part B of thermal-vacuum testing was initiated with testing at 35°C. Testing at 0°C was begun on December 17 and completed on December 29 after several problems* had been encountered and resolved. The spacecraft was then successfully subjected to 24 hours of testing at the 15°C level and 12 hours at the 35°C level; on December 31, 1965, the requirements for Part B of thermal-vacuum testing were completed.

5. Final Checks and Measurements

a. General

The spacecraft OT-3 testing program concluded with a series of final checks and measurements performed after the completion of environmental testing. When this portion of the program was completed, the spacecraft was inspected and made ready for shipment to the launch site. The final checks and measurements on spacecraft OT-3, which were initiated on January 3, 1966, and completed on January 12, included the following:

(1) A mechanical inspection, performed to ensure that all fasteners were tight, that all electrical connections were secure, and that the spacecraft was free from foreign material.

(2) An electrical test of systems operation (basically a repeat of the SEPET).

(3) Measurements of the spacecraft's magnetic dipole moments; final checks of the alignment of the TV cameras, the horizon-scanner pairs, the V-head attitude sensor, and the solar-aspect indicator; brightness-response and distortion and centering measurements for both TV cameras; final balancing and weighing of the spacecraft; a final dynamic spin test; and measurements of the spacecraft's moments of inertia.

The procedures used in the final alignment, in the final dynamic spin test, and in the final balancing were essentially the same as those used in the initial efforts. The alignment of the sensors on spacecraft OT-3 is illustrated in Figure II-35. (The values presented are those obtained from the final alignment procedures.)

*The problems encountered during thermal-vacuum testing and referenced herein are detailed, along with the associated solutions, in the History of Qualification Testing.

b. Final Alignment of Horizon-Scanner Pairs A and B

The positioning of the horizon scanners on the baseplate is shown in Figure II-35. On January 3, 1966, final alignment of the horizon scanners was performed. For each scanner, measurements were made of the displacement of the mechanical axis from the camera axis, the displacement of the optical axis from the camera axis, and the deviation between the mechanical and optical axes. These measurements are presented in Table II-12.

TABLE II-12. ORIENTATION OF SPACECRAFT OT-3 HORIZON SCANNERS

Horizon Scanner No.	1A	1B	2A	2B
Serial No.	109	112	113	108
Included Angle Between Optical Axis of Horizon Scanner and Associated Camera	62°43.5'	62°43'	62°43'	62°44'

c. Final Alignment of V-Head Attitude Sensor

The positioning of the two sensors which comprise the V-head attitude sensor can be noted in Figure II-35. On January 3, 1966, the angles which define the orientation of the V-head attitude sensor were measured. These measurements are presented in Table II-13.

TABLE II-13. ORIENTATION OF SPACECRAFT OT-3 V-HEAD ATTITUDE SENSOR

Parameter	"Up-Looking" Sensor	"Down-Looking" Sensor
Measured Angle Between Mechanical Axis and x-y Plane	50°07'	50°07'
Angle Between Optical Axis and Mechanical Axis*	-05'	+05'
Angle Between Optical Axis and x-y Plane	50°02'	50°12'

*Information supplied by manufacturer.

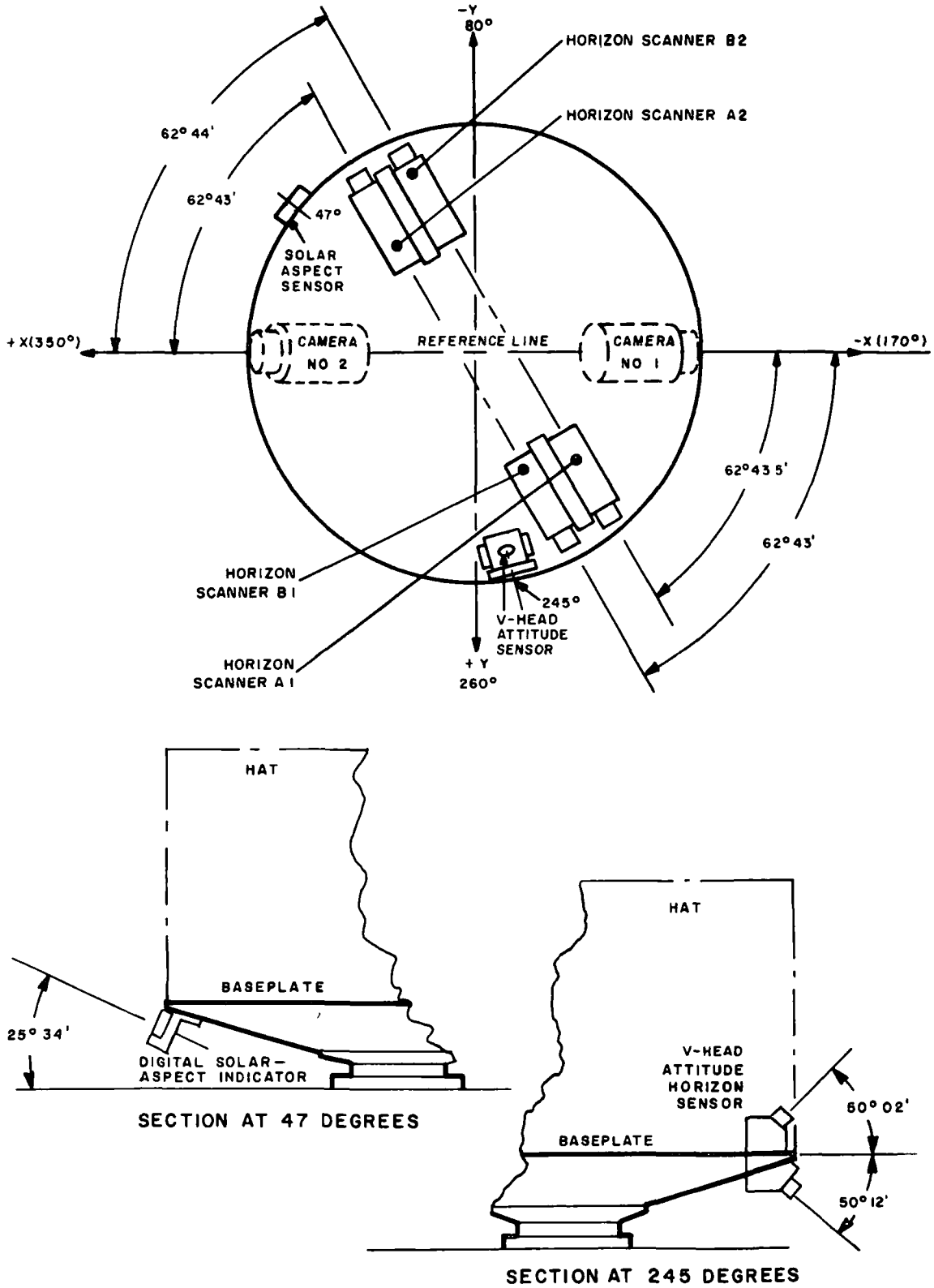


Figure II-35. Spacecraft OT-3 Sensor Alignment

d. Final Alignment of Solar-Aspect Indicator

The positioning of the solar-aspect indicator can be noted in Figure II-35. A knowledge of the exact orientation of the solar-aspect indicator is of great significance since the binary-code output from the indicator, including the satellite sun angle, is a function of the angle between the sun line and the normal to the front surface of the indicator.

On January 3, 1966, the angle between the mechanical axis of the solar-aspect indicator and the baseplate plane was measured and found to be 25°34'.

e. Final Alignment of TV Cameras

The positioning of the two TV cameras on spacecraft OT-3 can also be noted in Figure II-35. The values indicated in this illustration were obtained on January 3, 1966. The view axis of camera No. 1 was determined to be directed along the 170-degree radial in the plane of the baseplate and "downward" at an angle of 26°35' with respect to that plane. The view axis of camera No. 2 was directed along the 350-degree radial, and "upward" at an angle of 26°31'.

f. Measurements of Brightness Response, Spectral Response, Distortion, and Centering of TV Cameras

The measurements of the brightness response, spectral response, distortion, and centering of the spacecraft OT-3 TV cameras were performed on January 11 and 12, 1966.

The subcarrier frequency-deviation for each camera system was checked to determine the effects of length of operating time. These checks were performed at various levels of scene brightness, and the results are presented in Figures II-36 and II-37. Both cameras were then checked to determine the effects, in each direction of shutter motion, of variations in scene brightness upon subcarrier frequency. The results of these tests are shown in Figures II-38 and II-39.

The relative spectral response of each camera was also checked. The graphs presenting the results of these checks were included earlier in this document, in the description of the TV-picture subsystem (Figures II-11 and II-12).

Figures II-40 and II-41 are direct mode photographs taken with cameras No. 1 and 2, respectively, to determine whether any distortion was being introduced by the lens of the two cameras and to check the centering of the vidicon fiducial markings on the optical axis of each lens.

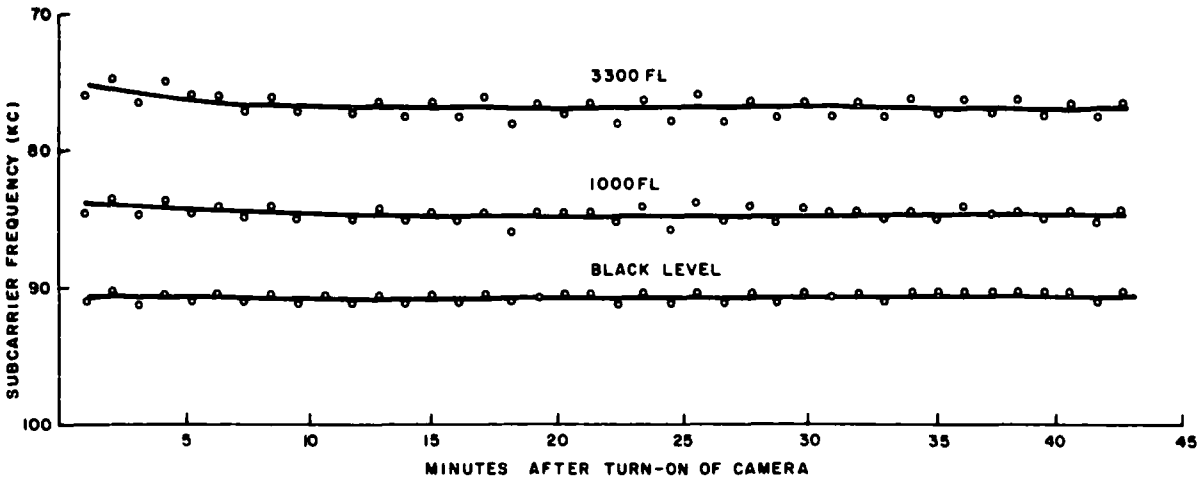


Figure II-36. Camera No. 1, Effects of Scene Brightness and Camera Operating Time on Subcarrier Frequency

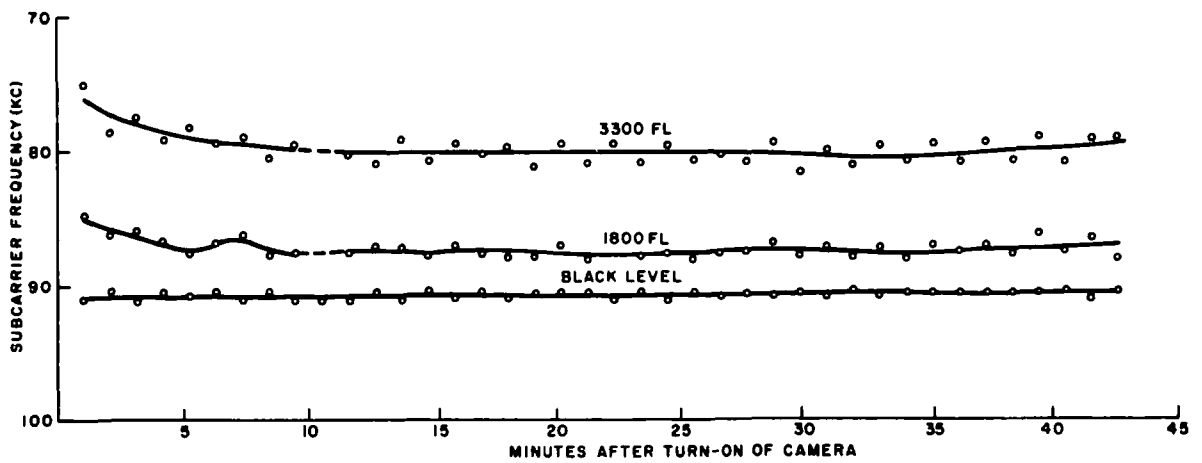


Figure II-37. Camera No. 2, Effects of Scene Brightness and Camera Operating Time on Subcarrier Frequency

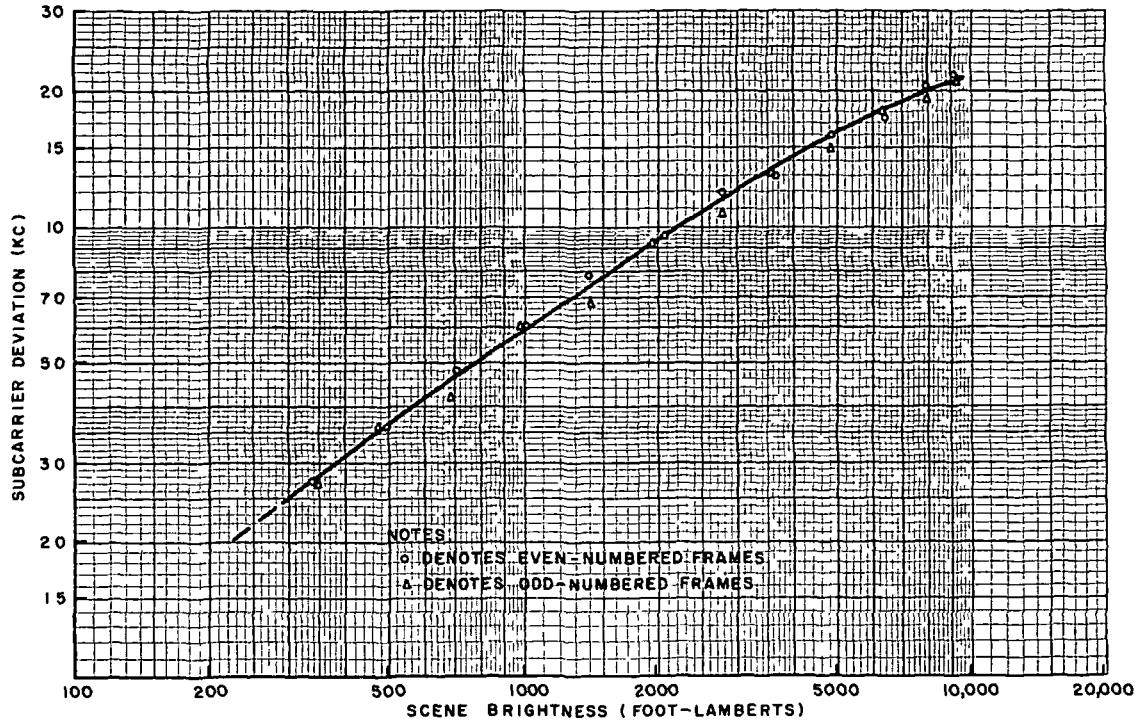


Figure II-38. Camera No. 1, Effects of Shutter-Motion Direction and Scene Brightness on Subcarrier Frequency

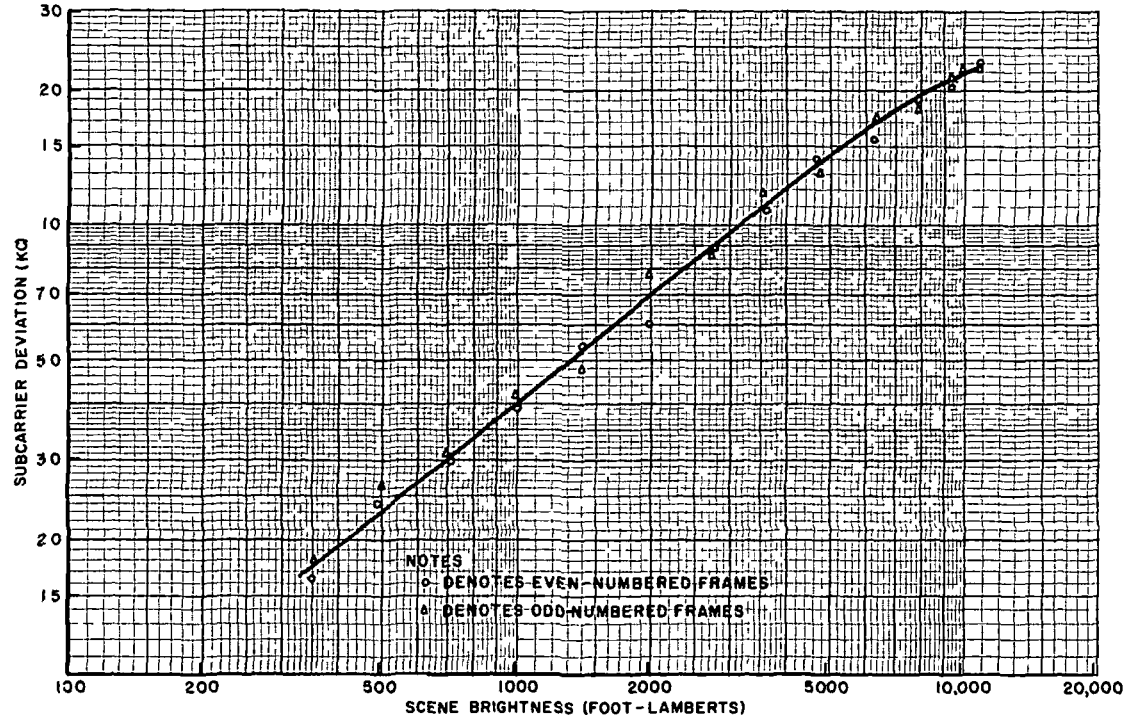


Figure II-39. Camera No. 2, Effects of Shutter-Motion Direction and Scene Brightness on Subcarrier Frequency

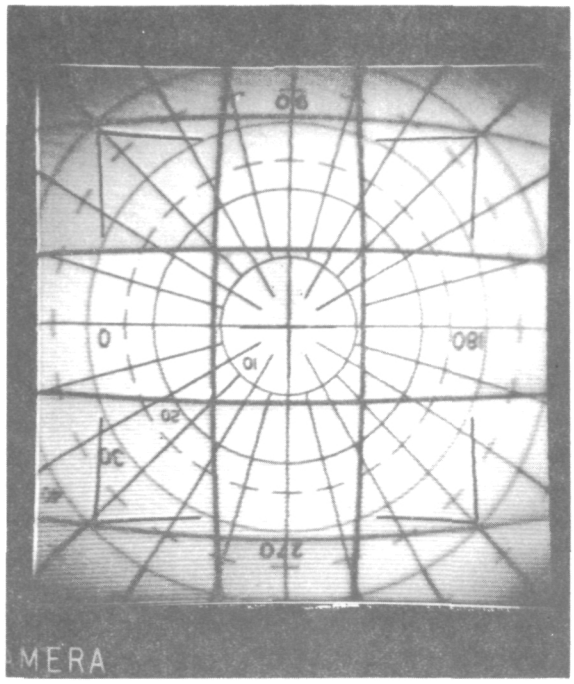


Figure II-40. Calibration Photograph; Camera No. 1, Direct Mode

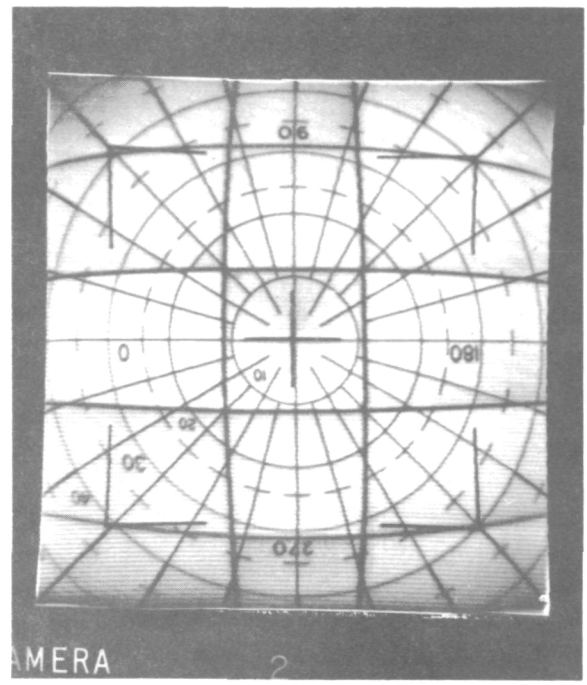


Figure II-41. Calibration Photograph; Camera No. 2, Direct Mode

g. Measurement of Magnetic Dipole Moments

On January 12, 1966, with spacecraft OT-3 operating in the standby mode, the residual magnetic moment was measured, then cancelled* by the addition of permanent magnets. The strength of the added magnets was 3.13 ampere-turns-meters^2.

After the magnets were added, the net magnetic moments for the various operating modes of the spacecraft were measured; the values obtained are listed in Table II-14. On that same date, the dipole moments generated by the MBC coil for each position of the MBC switch were measured. These measurements are listed in Table II-15, along with telemetered voltage indicative of each switch position.

For these procedures, the spherical dipole testing machine and the magnetometer were utilized.

h. Final Dynamic Spin Test

The final dynamic spin test was performed on January 10, 1966, with satisfactory results.

*To within the tolerance of state-of-the-art measurement apparatus. A small residual dipole moment remains and its effect in orbit is cancelled using the spacecraft's MBC coil.

TABLE II-14. MEASURED MAGNETIC-DIPOLE MOMENTS OF SPACECRAFT OT-3

Dipole	Operating Mode	Magnetic Moment (Amp.-Turns-Meters^2)
Spin-Axis	Standby	- 0.22
	Clock I Remote	+ 0.08
	Clock I Timing	- 0.01
	Clock II Remote	- 0.09
	Clock II Timing	
	Quarter-Orbit Magnetic Attitude-Control Coil	
	High Torque, Positive	+25.40
	High Torque, Negative	-26.90
	Low Torque, Positive	+10.80
	Low Torque, Negative	-11.00
	Playback I	+ 0.10
	Playback II	- 0.06
	Direct Camera No. 1	+ 0.15
	Direct Camera No. 2	- 0.05
Cross-Axis	Standby	+ 0 36
	Magnetic Spin-Control Coil	
	Positive Dipole	+ 6.10
	Negative Dipole	- 6 35

i. Final Balancing and Weighing of Spacecraft OT-3

Spacecraft OT-3 was first subjected to final balancing procedure on January 4, 1966.

- Baseplate: 306.0 gms at 329-degree radial
  122.0 gms at 75-degree radial
  115.0 gms at 28-degree radial
  22.5 gms at 194-degree radial
- Hat: 340.0 gms at 250-degree radial
  193.5 gms at 30-degree radial
  50.0 gms at 282-degree radial
  23.5 gms at 26-degree radial
  12.0 gms at 90-degree radial
  7.0 gms at 215-degree radial

TABLE II-15. MAGNETIC DIPOLE MOMENTS AND TELEMETERED VOLTAGES FOR THE MBC COIL

Switch Position	Selected Polarity**	Magnetic Moment (Amp.-Turns-Meters^2)	Telemetered Voltage On Recording Chart*** (Volts)
0 or 12	+	0	2.5
	-	0	
1	+	+1.97	0.5
	-	-1.96	
2*	+	+1.74	3.0
	-	-1.74	
3*	+	+1.50	0.9
	-	-1.50	
4*	+	+1.33	3.4
	-	-1.33	
5	+	+1.19	1.3
	-	-1.20	
6*	+	+1.00	3.8
	-	-1.00	
7*	+	+0.82	1.8
	-	-0.82	
8*	+	+0.60	4.3
	-	-0.60	
9*	+	+0.41	2.1
	-	-0.41	
10	+	+0.19	4.8
	-	-0.18	
11	+	0	0
	-	0	

*Magnetic moment is a calculated value.
**Recorded on telemetry channel 58, beacon I (positive polarity indicated by -0.5 volts, negative by -1.5 volts); or on channel 87, beacon II (positive polarity indicated by -3.3 volts; negative by -4.1 volts).
***Recorded on telemetry channel 59, beacon I, or channel 88, beacon II.

Because of the hybrid-matching circuit and new RF cable installed on the spacecraft* on January 7, it was necessary to repeat the final balancing procedures on January 11. At the conclusion of these test procedures, the balance weights present on the spacecraft were as follows.

- Baseplate: 306.0 gms at 329-degree radial
  122.0 gms at 75-degree radial
  115.0 gms at 28-degree radial
  85.0 gms at 245-degree radial
  6.0 gms at 120-degree radial
- Hat: 340.0 gms at 250-degree radial
  193.5 gms at 30-degree radial
  50.0 gms at 282-degree radial
  20.5 gms at 18-degree radial
  12.0 gms at 90-degree radial
  7.0 gms at 215-degree radial

After final balancing, the residual dynamic imbalance of the spacecraft was within the specified limits of 6 ounch-inches in each of the two planes.

On January 12, 1966, the final weight of the spacecraft was measured and was found to be 301.56 pounds without the despin weights, and the center of gravity of the spacecraft was found to be 10.44 inches above the separation plane.

f. Determination of Moment of Inertia

The spacecraft's moments of inertia were also determined on January 12. The data required for making these determinations was obtained by attaching the spacecraft to a bifilar suspension system, as shown in Figure II-42, and measuring the period of the resultant pendulum. Various configurations and attitudes were covered, as can be noted in Table II-16, which presents a summary of the mechanical and physical parameters of spacecraft OT-3.

D. HISTORY OF QUALIFICATION TESTING OF SPACECRAFT OT-3

a. Electrical Debugging and Initial Testing

The assembly of spacecraft OT-3 was initiated in August 1964 and completed in September, except for the V-head attitude sensor and the flight-model battery pack (as per test requirements, a spare battery pack was installed on the baseplate). Debugging was initiated at that time and continued through October and November.

*The conditions leading to the installation of this equipment are detailed in the "History of Qualification Testing" later in this section.

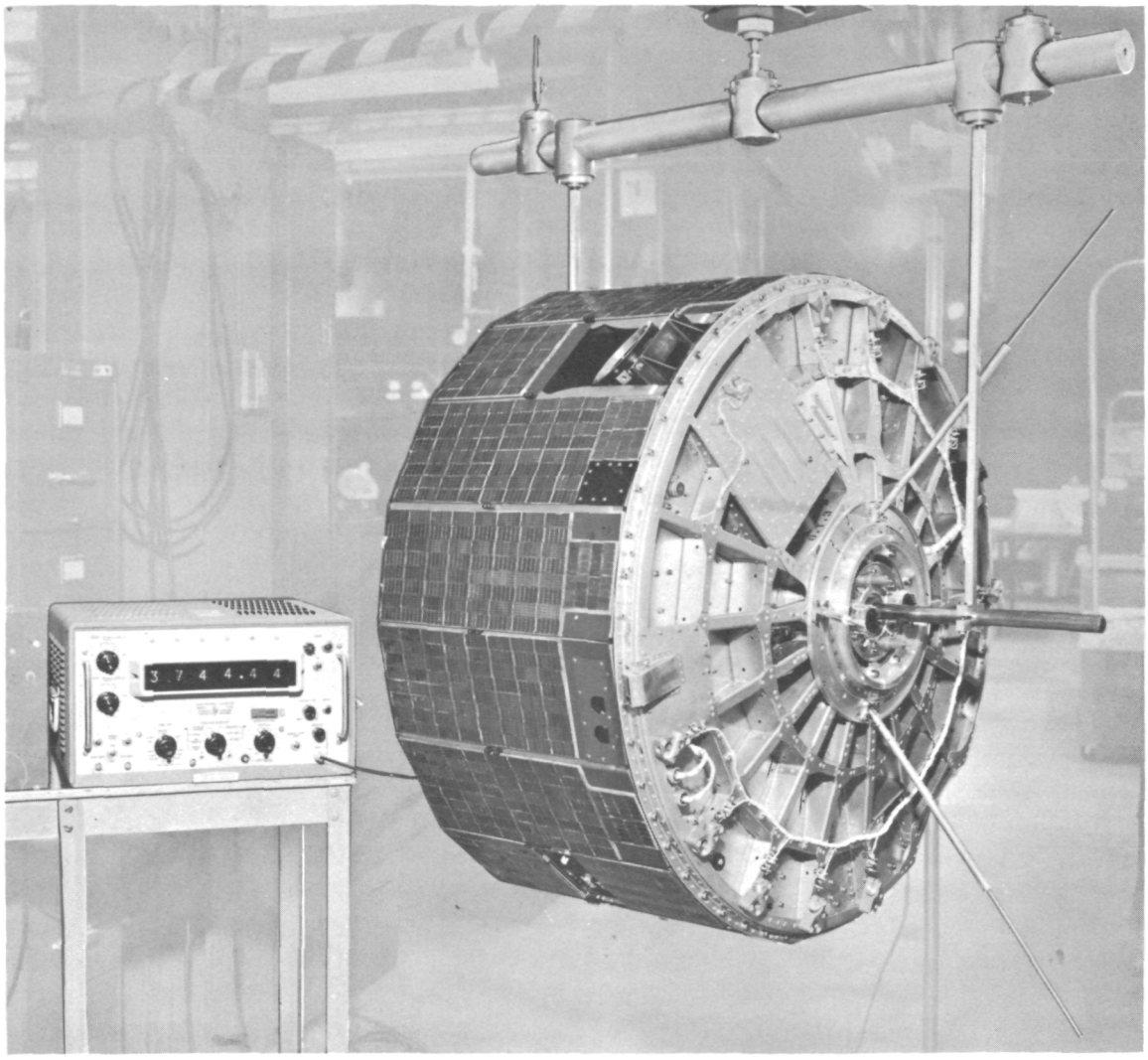


Figure II-42. Spacecraft OT-3 in Bifilar Suspension System for Moment-of-Inertia Tests

In the course of the debugging activity during November, tape transport No. 1 was observed to produce a higher than normal amount of flutter, necessitating the replacement of the transport with a qualified spare unit. In addition, it was noted that the video signal produced during operation of the cameras showed the effects of microphonics resulting from shutter operation. This condition was eliminated by replacement of the shutter assembly on each camera.

By November 23, debugging and the installation of all subsystem bracketry had been completed, and on November 24, a complete SEPET was initiated. This test was completed on December 1, with satisfactory results.

TABLE II-16. MECHANICAL AND PHYSICAL PARAMETERS OF SPACECRAFT OT-3

Parameters	With Despin Weights	Without Despin Weights
Spacecraft Weight	304.099 lbs	301.563 lbs
Moment-of-Inertia		
(a) About Spin Axis	166.740 lbs-in-sec^2	164.028 lbs-in-sec^2
(b) About Transverse Axis (Maximum)	117.926 lbs-in-sec^2	117.549 lbs-in-sec^2
(c) About Transverse Axis (Minimum)	112.454 lbs-in-sec^2	109.587 lbs-in-sec^2
(d) Ratio of (b)/(a)	70.7 percent	71.7 percent
(e) Location of Maximum Moment of Inertia	97° 36' (with keyway in separation ring at 90° and angles increasing clockwise with spacecraft viewed from hat)	96° 04'
Center of Gravity (above separation plane)	10.47 inches	10.44 inches
Dynamic Balance Residual dynamic imbalance (maximum)	98.86 oz-in^2 (0.00033 radian displacement between geometric and inertial axes)	
Static Balance Residual static imbalance (maximum)	9.40 oz-in (C.G. displacement of 0.0019 in)	

On December 2, preliminary alignment procedures were performed on the TV cameras, V-head attitude sensor, the horizon scanners, and the solar-aspect indicator. In addition, preliminary adjustments were made, on each camera system, of dark current, vertical and horizontal size, centering, brightness level, subcarrier deviation, electrical focus, and amplitude of the 10-kc picture time-delay signal.

The initial balancing procedures were performed on December 4; and on December 7 a receiver command sensitivity check and an RF-interference check were successfully performed using the outdoor tower facility. (To prevent the occurrence of electrical distortion during testing, this facility is constructed entirely of wood.) No discernible RF interference was present in either camera system, and the spacecraft was then prepared for the initial dynamic spin test. This test was successfully performed on December 8, verifying the proper operation of the components that initiate and control spin triggering of the cameras. By means of this test the effects of varying spin rate, attitude, and roll-angle error were checked. The special dynamic test facility designed and first used on the TIROS IX program was also used for spacecraft OT-3.

At this point spacecraft OT-3 had completed the initial debugging, checks, and measurements, and was ready for the initiation of environmental qualification testing.

b Environmental Testing

Environmental testing of spacecraft OT-3 was initiated on December 10, 1964, with Part A of vibration testing. (The test levels and types of vibration covered are listed in Table II-9.) The tests were successfully completed and no anomalies were observed. However, an inspection after completion of the tests revealed loose hardware in the tape recorder power control unit (Serial No. 3-322-12). The unit was removed from the spacecraft and a loose screw was found; the screw was installed properly and tightened, and the unit was reinstalled on the baseplate. An abbreviated SEPET was successfully performed on December 11, and on the following day Part A of thermal-vacuum testing was initiated. (See Table II-10 for the temperature levels and test durations covered.) The temperature was stabilized at the 50°C level on December 13, but after only 5.5 hours at this level the absence of playback video from Side 1 was observed. The temperature was lowered to 25°C but the condition was not alleviated. The spacecraft was removed from the chamber on December 14, and tape transport No. 1 and the associated R-2 box were removed from the spacecraft for examination.

An examination revealed that the tape on transport No. 1 had been completely transferred to one reel. The problem was traced to a defective record-play-back motor-reversing relay in the power control unit (the R-2 box, Serial No. 12). The

relay had failed to reverse the direction of the drive motor when the end-to-tape was sensed in the record direction. Thus, when the next playback operation was started, the tape continued to travel in the record direction until it became completely unwound from one reel and transferred to the other. A new, flight-qualified power control unit (Serial No. 13) was installed, and thermal-vacuum testing at the 50°C level was resumed.

On December 17, there was no response to a command for switching time base generators; the test was continued using the time base generator which was in operation at the time of the malfunction. On December 18, telemetry commutator No. 2 stopped operating. The temperature was then lowered to 0°C. On December 20, the camera control unit (Serial No. 3-621-03) and the signal conditioner package (Serial No. 3-925-03; Commutators Serial Nos. 304 and 299) were removed from the spacecraft for examination.

The malfunction in the camera control unit was isolated to CR9, a IN964B, 13-volt zener reference diode which determines the operating level of the power relay driver circuits. Further analysis by the manufacturer revealed evidence of "channelling," a failure mode which results in concentration of current flow in a minute portion of the junction area. As such, the failure mode was classified as very rare and one which occurs on a random basis. The diode was replaced and the camera control unit, after being requalified, was reinstalled on spacecraft OT-3 on December 22.

Telemetry commutator No. 2 was examined by the manufacturer. Series input resistor, R224, in the power inverter was found to be mechanically intact, but upon X-ray examination an open condition was revealed. A new, qualified commutator (Serial No. 305) was integrated with the signal conditioner package, and the package was re-installed on spacecraft OT-3 on December 24. Thermal-vacuum testing was resumed on December 26 at the 50°C level. On December 28, however, testing was halted when a similar failure occurred with telemetry commutator No. 1. The signal conditioner package was removed from the spacecraft and the telemetry commutator was examined, revealing an internal open condition with series input resistor, R224, and an internal shorted condition with series diode, D209.

It was decided to modify the commutators by bypassing the questionable resistor and series diode and adding the required series resistance external to the units. Accordingly, the signal conditioner package with reworked commutators and associated series resistors was reinstalled on the spacecraft on January 8, 1965.

Thermal-vacuum testing was resumed on January 10, at the 50°C level. On January 13, the test temperature was lowered to 0°C, and on January 15 the Part A thermal-vacuum test requirements for spacecraft OT-3 were successfully completed.

On January 25, an abbreviated SEPET was performed with satisfactory results. On February 12, 1964, spacecraft OT-3 was formally placed in the "hold" status by NASA directive.

Throughout the "hold" period, the spacecraft was subjected to Go-No/Go tests at least once each month to verify normal operation. During testing on September 2, tape recorder No. 1 failed to respond to a playback command. Power was then switched to recorder No. 2, which functioned normally. Tape recorder No. 1 was removed from the spacecraft for further examination of the power control unit. (The Go-No/Go test during October was performed without tape recorder No. 1). As a result of the study of the tape recorder failure of September 2, the following modifications were effected:

- The drive belt, end-of-tape microswitch, and magnetic tape was replaced on tape transport No. 1.

- The transistors in the end-of-tape amplifiers in both power control unit No. 1 and in the spare unit were replaced because the study indicated the possibility of overstressing during evaluation tests conducted since the time of the failure.

On October 18, 1965, the "hold" on spacecraft OT-3 was formally rescinded by NASA directive.

On November 18 and 19, rf-interference tests and command-sensitivity tests were performed at AED with the spacecraft mounted on the NASA-furnished model of the third-stage launch vehicle. Test results indicated the necessity for a system of 136-Mc bandpass filters and 3-db attenuators in the RF coupling system between the beacon transmitters and the diplexer to afford adequate isolation during the launch phase and to minimize spurious generation of rf interference and the coupling of such interference to the command receiver.

The flight-model battery pack was installed on the spacecraft on November 22. From November 26 to 28, an abbreviated SEPET and selected portions of the SEPET were performed with satisfactory results.

On November 29 and 30, the spacecraft was subjected to temperature testing (24 hours at 0°C and 24 hours at 35°C) utilizing the Tenney chamber.

On December 2, a SEPET was initiated. The spacecraft was also taken outdoors for a solar-cell array test.

At a meeting of the Environmental Test Committee on December 2, the Part B requirements for both thermal-vacuum and vibration testing were increased to the levels given in Table II-10.

On December 3, the one-year timing devices, which were not to be utilized on the ESSA 1 mission, were removed from the baseplate and weights were substituted to preserve the balance and inertia characteristics of the spacecraft. (The timing devices had not been used on TIROS missions since the TIROS IX program; on TIROS IX, however, the devices had simply been disconnected.)

On December 9, subsystem tests were completed on the beacon transmitter assembly and modified RF coupling system, and the units were mounted on the spacecraft. The SEPET initiated on December 2 was successfully completed on December 9 and additional rf interference tests were performed in connection with the model of the third-stage launch vehicle (X-258).

Based on the satisfactory results of these tests*, it was anticipated that the modification would in fact, improve the isolation of the beacon transmitters, and thus reduce their spurious radiation to the command receiver.

On December 11, Part B of vibration testing, as modified by the Environmental Test Committee on December 2, was performed with satisfactory results. On the following day, an abbreviated SEPET and a Go-No/Go test were successfully performed.

Part B of thermal-vacuum testing of spacecraft OT-3 was initiated on December 14, at the 35°C level. Two days of testing at this temperature were successfully completed, and the test level was lowered to 0°C. After 24 hours at this level, however, testing was halted because of a spurious signal from TV transmitter No. 2 for approximately 1.5 minutes after each turn-on. The spacecraft was removed from the chamber, and the transmitter was removed.

A qualified spare transmitter was installed on the spacecraft and thermal-vacuum testing was re-initiated at the 0°C level. After approximately 14 hours at 0°C, testing was halted when horizon scanner B2 failed to produce an output when exposed to a light source. Subsequent tests indicated that all other scanners were normal. A check was made of the test console interface and horizon scanner target power source, which established the continuity and current levels of the test lamps.

The spacecraft temperature was raised to ambient and tests indicated normal operation. The spacecraft was removed from the vacuum chamber and tested in the Tenney box while the temperature was lowered to 0°C. The scanner outputs were continuously monitored and, at a discrete temperature, the output of scanner 2B disappeared and only the d-c bias remained.

*A detailed discussion of the system modifications and the RF-interference tests performed is presented in Appendix B.

The scanner assembly was removed from the spacecraft and the interface between the threshold amplifier and the scanner was tested. The apparent intermittent operation was repeated in the Tenny box and determined to be within the horizon scanner.

Two spare horizon scanners were tested on December 12, and both were found to be acceptable. An additional bench test for field-of-view alignment, and Tenney box tests at "hot" and "cold" temperature levels were conducted on the spare flight-qualified unit (Serial No. 3-921-108). A thrust axis vibration test was also performed to establish additional confidence in the mechanical integrity of the unit. The results of this test were satisfactory.

Horizon scanner Serial No. 3-921-108 was then mounted on the spacecraft and electrically tested to establish proper performance with interface units. The spacecraft was returned to thermal-vacuum testing on December 27, 1965.

Testing was performed at 0° C for 48 hours, 15° C for 24 hours, and 35° C for 12 hours, and completed on December 31 with satisfactory results. This satisfied the environmental test requirements on the spacecraft OT-3 test program. Table II-17 presents a summary of the temperature levels experienced during thermal-vacuum testing of spacecraft OT-3 and the test durations at each level.

On January 1 and 2 a SEPET was successfully performed on spacecraft OT-3; and on January 3 final alignment of the TV cameras, solar-aspect indicator, V-head attitude scanner and horizon scanners was performed.

On January 4, final balance procedures were performed, and the spacecraft's moments of inertia were measured. During a command sensitivity test on this date, command receiver No. 2 was found to be approximately 14 db less sensitive than receiver No. 1. Upon examination, the RF input cable of receiver No. 2 was found to have an intermittent open circuit in the shield connection at one of the connectors. A hybrid matching circuit and a new receiver cable were installed, necessitating the relocation of the notch filter. This effort was completed on January 7. On the same date, the despin weights and cables were installed, and, because of the modifications, it was decided to perform additional qualification tests. The spacecraft was subjected to random vibration in the thrust axis on January 7. After examination showed no untoward effects of vibration testing, a Go-No/Go test was successfully performed. The spacecraft was then subjected to temperature testing in the Tenney chamber. With the successful performance of the additional qualification tests, the final balancing procedures were repeated on January 11.

TABLE II-17. SUMMARY OF THERMAL-VACUUM TESTS OF SPACECRAFT OT-3

Temperature	Test Number							
	1	2	3	4	5	6	7	Total
50° C	10 hrs	60 hrs	19 hrs	26 hrs	-	-	-	115 hrs
35° C	-	-	-	-	28 hrs	-	12 hrs	40 hrs
15° C	-	-	-	-	-	-	26 hrs	26 hrs
0° C	-	23 hrs	-	74 hrs	9 hrs	20 hrs	50 hrs	156 hrs
							Total	337 hrs

Camera distortion and centering and camera brightness tests were completed on January 12, and on the same date measurements were made of the spacecraft's weight, center of gravity, and moments of inertia. An abbreviated SEPET was performed on January 13 and new despin weights were installed. On January 14 the spin-up rockets were installed; final inspection of the spacecraft was then performed and the spacecraft was placed in its shipping container. On January 15, at NASA's direction, the spacecraft was shipped from AED to the launch site at the Eastern Test Range.

PART III. GROUND-STATION EQUIPMENT

SECTION I. INTRODUCTION

A. GENERAL

Ground operations for the ESSA 1 satellite system are performed and coordinated by a ground complex consisting of the following installations: the TIROS Technical Control Center (TTCC)located at the Goddard Space Flight Center in Greenbelt, Maryland, two primary Command and Data Acquisition (CDA) stations, one located at Wallops Island, Virginia, and another Fairbanks, Alaska; a secondary CDA station located at the RCA Space Center near Princeton, N..J.; and selected stations of the NASA Minitrack Network. The CDA stations perform the command and control functions for the satellite system, and receive, process, and record all data transmitted from the satellite. The two primary CDA stations and the secondary station are functionally identical and differ only in physical arrangement, housing, and minor equipment variations. (The equipment variations are described in detail in the succeeding paragraphs.)

The precision-tracking capabilities of the selected Minitrack stations are used for accurately determining the satellite's orbital parameters. The ESSA 1 system does not include any satellite-borne IR experiments. Consequently, the IR equipment located at the CDA stations for receiving, storing, and handling IR data is neither noted nor discussed in the following descriptions.

B. PHYSICAL DESCRIPTION

Most of the components of the CDA ground stations are mounted on roll-out assemblies and housed in vertical racks which have an over-all height of 65-3/8 inches. Each roll-out assembly can be fully extended from the rack and can be locked in the extended position with locking devices provided on each rail. Two knurled, adjustable, pawl fasteners, mounted through the front panel of each unit, secure the roll-out assembly to the rack in the retracted position. Each roll-out assembly consists of a front panel and two vertically mounted chassis arranged so that the electron tubes face inward and the wiring outward, thereby facilitating maintenance. Cooling is accomplished by twin squirrel-cage blowers located at the bottom of each rack, which force air up through the rack to be exhausted through a louvered vent at the top. Reuseable air cleaners are provided at the air intake vents.

Table III-1 lists the equipment racks by number and name, and indicates the major components located in each rack. Figure III-1 shows the rack arrangement in each of the CDA stations.

TABLE III-1. CDA GROUND STATION EQUIPMENT

Rack No.	Rack Name	Equipment	Notes
1	Command Control Program A	Remote Picture Time Set	Not used for ESSA 1
		Digital Command Programmer	---
		Command Programmer	---
		Control Tone Generator	---
2	Command Control Timing	WWV Receiver	---
		Master Clock	---
		Master Clock Alarm Unit	---
		Program Selector and Power Control Unit	---
		WWV Comparator	---
		Frequency Standard	---
		300-Volt Power Supply	---
		Relay Power Supply	---
3	Command Control Program B	(Same as Rack 1)	---
4	Video Monitor	Display and Video Amplifier	---
		Sawtooth and Deflection Amplifier	---

TABLE III-1. CDA GROUND STATION EQUIPMENT (Continued)

Rack No.	Rack Name	Equipment	Notes
4 (Cont'd)		Sync Separator and Picture Delay Display Unit	---
		FM Demodulator - TV	---
		High-Voltage Power Supply	---
5	Video Control and Power	Monitor Oscilloscope	---
		Tape and Computer Control	---
		Monitor Control	---
		32-Volt Power Supply	---
		300-Volt Power Supply	---
		Relay Power Supply	---
6	Sun Angle Computer Computer	Index Generator and Readout Control	Not used for ESSA 1
		Sun Angle Computer and Elapsed Time Indicator	
		Control Logic	
		Arithmetic Units	
		Power Supply (-26V, -6.5V, 6.5V)	
7	Beacon Receiver and Attitude Recorder	Telemetry Receiver (2)	---
		Attitude Recorder (Sanborn No. 1)	---
		+6-Volt Power Supply	---
		-12-Volt Power Supply	---

TABLE III-1. CDA GROUND STATION EQUIPMENT (Continued)

Rack No.	Rack Name	Equipment	Notes
8	Tape Recorder No. 1	TV Tape Recorder	---
9	Tape Recorder No. 2	TV (and IR) Tape Recorder	IR Channels not used for ESSA 1
10	TV and IR Receivers	Television Receivers (2)	---
		Infrared Receivers (2)	Not used for ESSA 1
		Diversity Combiner (2)	---
		+6-Volt Power Supply	---
		300-Volt Power Supply	---
11	Calibration	FM Signal Generator	---
		Patch Panel	---
		Calibration	---
		Transmitter Control Panel	---
		Frequency Counter	---
		Audio Oscillator	---
		Audio Amplifier	---
12	Attitude Readout Facility	WWV Monitor Oscilloscope	---
		Digital Time-Measuring Device	---
		Tape Punch	---
		Attitude Pulse Selector	---

TABLE III-1. CDA GROUND STATION EQUIPMENT (Continued)

Rack No.	Rack Name	Equipment	Notes
13	Command Transmitter A and Transmitter Selector	Command Transmitter and Associated Power Supply	Racks 14 and 15 are not located with other racks, but are instead housed with the antennas. Racks 14 and 15 at Fairbanks are not AED-furnished equipment
14	Command Transmitter B	Command Transmitter and Associated Power Supply	
15	Telemetry Receiver and Control	Telemetry Receiver (2) Telemetry Receiver Control Antenna Multicoupler (2) Frequency Converters	Located at Wallops Island and Fairbanks only
16	Events Recorder	Events Recorder RF Monitor Magnetic Tape Degasser	Located at Wallops Island and Fairbanks only.
17	Tape Recorder Electronics	Tape Recorder Electronics	Located at Wallops Island only.
18	Quick Look Demodulator	Oscilloscope Ring Modulator Counter Demultiplexer (2)	Not used for ESSA 1

TABLE III-1. CDA GROUND STATION EQUIPMENT (Continued)

Rack No.	Rack Name	Equipment	Notes
18 (Con't)		Tape Error Compensator	Not used for ESSA 1
		Mixer and Time Separator Unit	
		Paper Tape Recorder	
		D-C Power Supply (2)	
19	IR Quick Look Readout	IR Buffer	Not used on ESSA 1
		Digital Time Measuring Device	
		Tape Punch	
20	AGC Calibrator	Sanborn Recorder (4-channel)	---
		AGC Calibrator	---
21	Telemetry Recorder	Telemetry Recorder Sanborn No. 2 (2-channel)	---
		Beacon Data Control Unit	---
		Subcarrier Discriminators (2)	---
		Telemetry Calibrator	---
		+28-Volt Power Supply	---
22	SCAMA Phone Link		

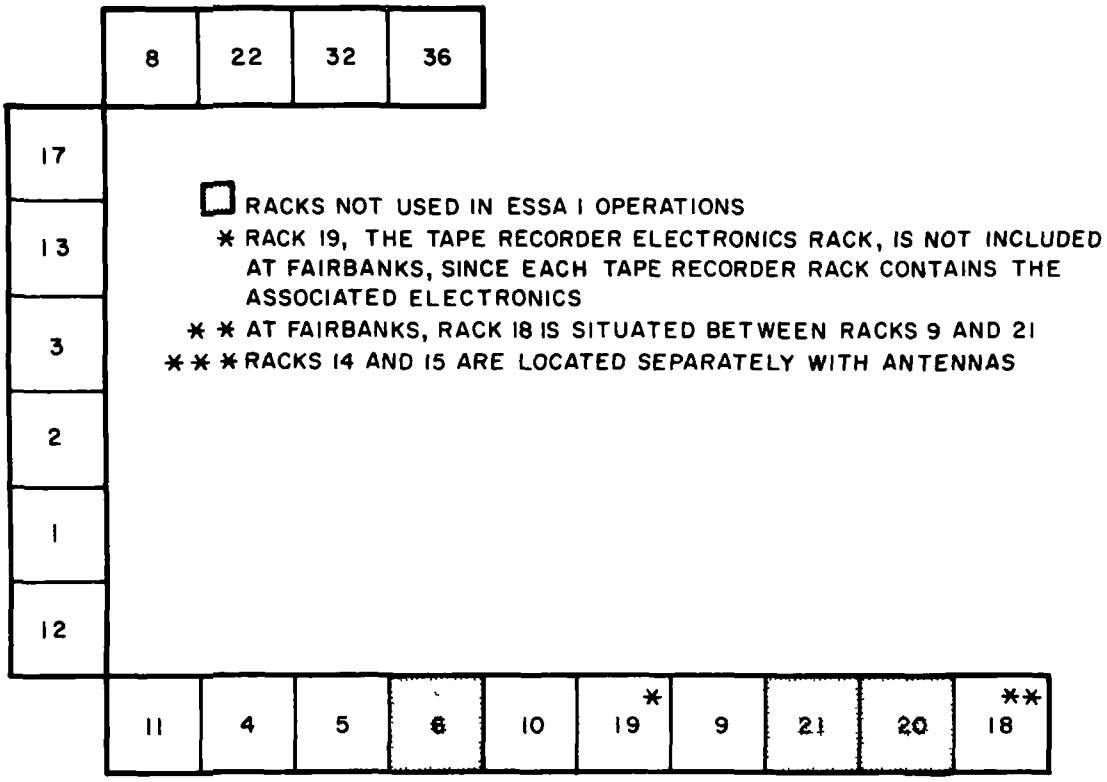


FIGURE III – 1A. RACK LOCATIONS AT WALLOPS ISLAND AND FAIRBANKS

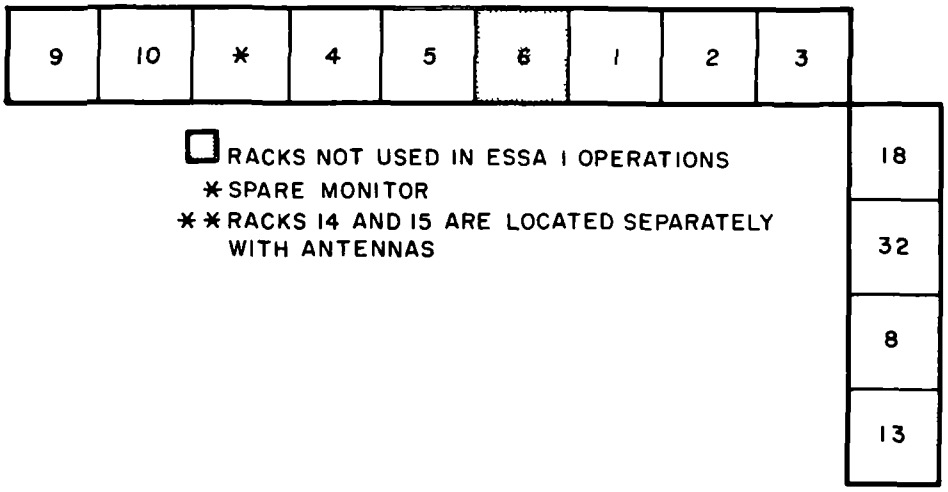


FIGURE III-1B. RACK LOCATIONS AT AED

Figure III-1. ESSA 1 Equipment Layout at CDA Stations

SECTION II. GROUND-STATION OPERATION

A. GENERAL

The basic functions of the CDA ground stations are: (1) to transmit radio command signals to the satellite, programming it for operation and data transmission; (2) to receive the radio signals carrying the television and other data necessary for orientation of the satellite and evaluation of satellite performance; (3) to process, record, and label the data received.

Functionally, the ground station equipment is arranged into six major subsystems, as follows: master timing equipment, programming equipment, TV receiving equipment, data processing equipment, TV display equipment, and beacon data receiving and recording equipment. The master timing equipment, which is synchronized with radio station WWV, provides the time standard for both satellite and ground stations, and generates alarm signals that initiate various phases of the automatically transmitted command program. The programming subsystem incorporates equipment that provides for the setting up of the command program prior to the satellite pass, and for the generation and transmission of the command signals to the satellite in the proper sequence. The TV receiving equipment provides for polarization-diversity reception of the satellite TV transmissions, while the TV display equipment demodulates the TV signals and processes the video data for kinescope presentation and film recording. The data-processing equipment provides a unique identification index for each TV picture received and also provides for the demodulation and signal processing of picture-delay data transmitted with each TV picture. The attitude, telemetry, and command-verification signals which are transmitted to the ground station over the satellite's two beacon transmitters are received, processed, and recorded by the components of the beacon data receiving and recording subsystem.

In addition to the subsystems listed above, each ground station is equipped with two Ampex RF 100 tape recorders used for permanent recording, on magnetic tape, of all satellite data transmissions, and an Esterline-Angus Model AW, 20-channel, events recorder which provides a permanent record of all major events occurring during a satellite pass. Each ground station is also equipped with calibrator unit and other test equipment necessary for checkout and maintenance operations on the major subsystems.

Figure III-2 contains a functional block diagram of the CDA ground-station equipment. Complete descriptions of the ground-station subsystems are given in the remaining paragraphs of this section.

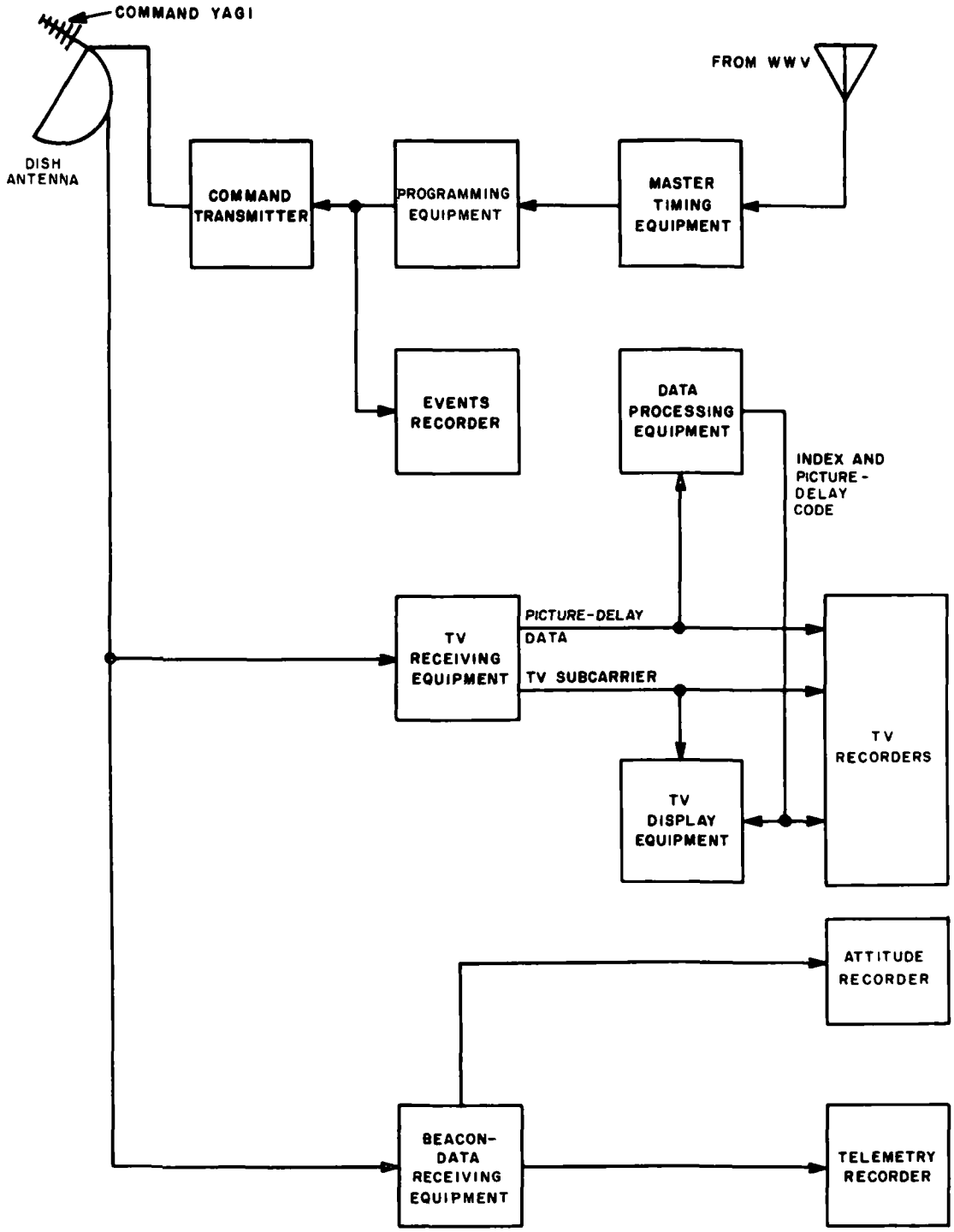


Figure III-2. CDA Ground Station Equipment, Block Diagram

B. OPERATION AT A CDA STATION DURING A TYPICAL SATELLITE PASS

Prior to the actual pass of the satellite over the CDA ground station, the selected command program, as specified in directions from TTCC, is set up on each of two analog programmers and two digital programmers in the programming and control equipment. The initiation times for the various program sequences are specified in the directions from TTCC and are set into the master timing equipment, which has been previously synchronized with timing signals received from radio station WWV.

When the satellite comes within ground-station range, the antenna is directed along the pre-determined tracking path. The master timing equipment alarms at the selected times, causing the programming equipment to initiate first the proper satellite address, and then the command sequences, in turn. Individual satellite commands, represented by audio-frequency tones (PCM or FSK) modulating the command transmitter, or a by frequency-shift-keyed tone, are transmitted to the satellite via the ground-station antenna. Indications of the satellite commands are recorded on the 20-channel events recorder to provide a permanent record of command initiation times and durations.

Upon receipt of the proper address and command tones, the satellite starts transmitting TV data to the CDA station. The TV data is received by the TV receiving equipment, which consists of two TV receivers connected in polarization diversity. The TV subcarrier is separated from the received signal and routed to the TV display equipment, where the video is demodulated from the subcarrier and the properly synchronized deflection voltages are generated. The resultant TV picture is displayed on a kinescope and automatically photographed. The TV subcarrier is also routed to the TV recording equipment and recorded permanently on magnetic tape.

Picture-delay data (i.e., data pertaining to the time interval between the generation of a picture request and actual shutter actuation) required for determination of the real time of the TV pictures is also included on the TV carrier. This data is filtered from the carrier, and applied to the data processing equipment for demodulation and processing. The picture-delay code from the data processing equipment is applied to the TV display equipment and presented as a black and white bar on the edge of the kinescope display. The picture-delay data is also recorded on separate channels of the TV recorders.

The data processing equipment also generates an index, consisting of a picture frame number, camera-source information, and the mode of satellite operation (i.e., direct-picture or playback) for each TV picture received. The indexing information is applied to the TV display equipment, where it is displayed on the front panel of the kinescope for photographing with the TV picture, and is also recorded permanently on the TV recording equipment.

Throughout the satellite pass, spin-axis-attitude information is continuously transmitted on subcarriers of both satellite beacon transmitters. This information is demodulated

from the subcarriers in the beacon data receiving equipment and is recorded on a 2-channel Sanborn paper-chart recorder, designated as the "attitude recorder". Two other subcarriers of the beacon transmitters are modulated, at various times during the satellite pass, with houskeeping-telemetry, solar-aspect, camera-triggering, command-verification, and back-up attitude data. These subcarriers are also demodulated in the beacon data receiving equipment and the resulting information is displayed on a second Sanborn recorder assigned to the telemetry transmissions and designated as the telemetry recorder.

At any time during the satellite pass, but normally towards the end of satellite-to-ground contact, any of the auxiliary commands (fire spin-up rockets, beacon turn-off, command telemetry, etc.) can be initiated by the ground station command and control equipment and transmitted to the satellite.

At the conclusion of the satellite pass, the data taken is arranged in a systemized format and transmitted to agencies in the ESSA 1 ground complex for evaluation.

SECTION III. FUNCTIONAL DESCRIPTIONS OF GROUND-STATION EQUIPMENT

A. MASTER TIMING EQUIPMENT

1. General

A block diagram of the master timing equipment for the CDA ground station is shown in Figure III-3. The equipment, which consists of a frequency standard, a master clock, a master clock alarm unit, a WWV comparator, and a WWV receiver, provides the time standard for both the ground station and the satellite, and generates the alarm signals that initiate the various sequences of a program during automatic programming operation. The timing pulses, a 2-second pulse occurring every 10 seconds, and the alarm signals are applied to both command programmer A in rack 1 and command programmer B in rack 3. In addition, a 100-pps pulse train, generated in the master clock, is applied as the time standard for the digital command programmers in both rack 1 and rack 3.

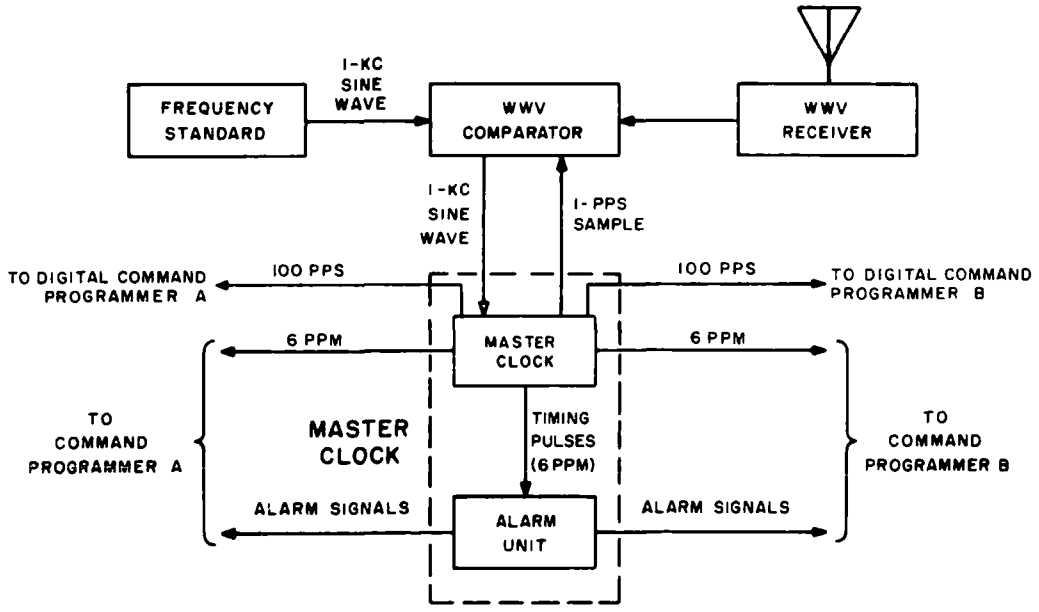


Figure III-3. Timing Equipment, Block Diagram

2. Frequency Standard

A Lavoie, Model LA-90, 5-megacycle frequency standard is used as the standard for the timing equipment. The unit utilizes a 1-megacycle, plano-convex crystal, operating in the fifth overtone mode, as the basic frequency source, and has a frequency stability rating of 1 part in 10^8 per day. Temperature compensation is obtained by using a crystal oven, whose temperature is controlled by phase comparison of two 2-megacycle signals, one of which is derived from the basic 5-megacycle oscillator, the other from a 2-megacycle crystal having a radical frequency-versus-temperature characteristic. The resultant output of this comparison is used as an extremely sensitive control for proportionate oven-heater power, thereby maintaining stable frequency output.

The basic 5-megacycle frequency is reduced to 1-megacycle, then to 100 kilocycles by regenerative frequency dividers. This 100-kilocycle frequency is further reduced to 10 kilocycles and 1 kilocycle, respectively, with two phantastron-type frequency dividers. The dividers in this series are isolated with cathode-follower stages at their respective outputs. The 1-kilocycle sine-wave output is applied to the WWV comparator.

3 WWV Comparator and WWV Receiver

The WWV compartor (a Hermes Model 222) and the WWV receiver (a Beckman/ Berkeley Model 905) permit the timing system to be synchronized with radio station WWV. The 1-kilocycle sine wave from the frequency standard is applied to the rotor of a phase-shifting resolver in the WWV comparator. This resolver is so designed that one complete revolution will shift the phase of the input signal 360 degrees. The 1-kilocycle, sine-wave output of the resolver is applied to the master clock unit, where it is converted to a 1 pps signal used to drive the master clock.

A 1-pps output from the master clock unit is coupled back to the WWV comparator and delayed the proper number of milliseconds to offset the propagation delay from the WWV transmitter (i.e., 1 millisecond for each 186 miles). The pulse is then used to trigger a 5-millisecond pulse which, in turn, is summed in a resistive network with the 1-second tick transmitted by WWV. (The 1-second tick consists of 5 cycles of a 1000-cycle sine wave.) The phase-shifting resolver may then be adjusted to bring the 5-millisecond pulse, and thus the 1-pulse-per-second clock pulse, in exact coincidence with the 1-second tone burst from WWV by observation of the signals on an oscilloscope.

4. Master Clock Unit

The master clock unit provides the fundamental timing pulses for operation of the ground station equipment and, when properly set, displays Greenwich Mean Time in hours, minutes, seconds, and milliseconds. The unit receives a 1-kc sinewave input from the WWV comparator and, by frequency division, produces synchronized rectangular output pulses of 100 pps (to the digital command programmers), 1 pps (to the WWV comparator), and 6 ppm (to the master clock programmer). Cascaded Burroughs DC-106A decade counters are used to perform the frequency division and counting functions and to present the time display. Shaping circuits are provided to change the sinewave input into a narrow rectangular pulse suitable for driving the counting chain. The 6-ppm pulse operates a relay within the master clock unit, which supplies, through its contacts, a driving voltage to operate the stepping switches in the alarm unit and the command programmer. The counting circuits are automatically reset to zero seconds whenever the accumulated count reaches 24 hours. A manual RESET button is also provided, as is a SET-RUN switch is used to disable the counter so that it may be set.

5. Master Clock Alarm Unit

The master clock alarm unit is used to generate the six alarm signals required to initiate the command sequences in each command programmer. It consists of six independent alarm circuits that can be set manually in hours, minutes, and 10-second steps, for a maximum total elapsed time of 24 hours. Stepping switches in each of the alarm circuits are driven by a 2-second pulse from the master clock every 10 seconds. Four time-setting selector switches are provided on the front panel of the alarm unit for each of the six alarm circuits. These switches, representing tens of seconds, minutes, tens of minutes, and hours, are adjusted to the time at which the alarm is required. Each point on the switch is wired to a corresponding terminal on the stepping switches and, when the contact wipers of the stepping switches and those of the time-setting selector switches coincide, a series circuit is set up and an 8-second alarm pulse is sent to the applicable command programmer.

6. Digital Time-Marker Equipment

The digital time-marker equipment allows correlation of recorded data with real time. As shown in Figure III-4, several types of time codes are generated by an EECO Model ZA34538 time-code generator/reader unit, which is driven by the basic 1-kc clock frequency and a 1-pps synchronization signal from the master clock unit. One of these time codes (in the "Generate" mode) is a modulated 1-kc output to channel 4 of Ampex magnetic tape recorder No. 1. On recorder playback, the reproduced output of this channel is fed back to the time-code generator/reader unit (in the "Reproduce" mode), which then regenerates the other codes based on the recorded time.

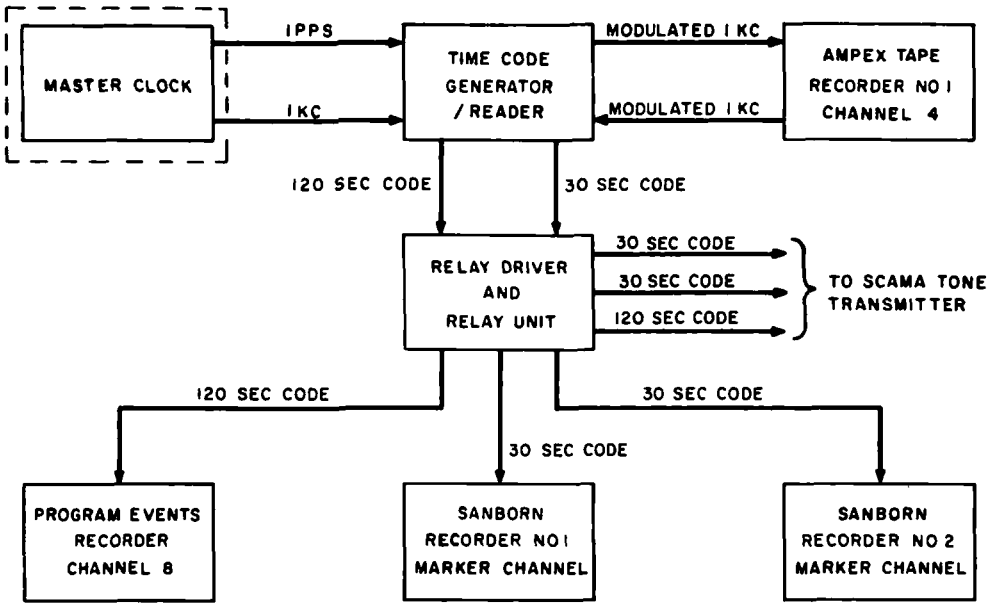
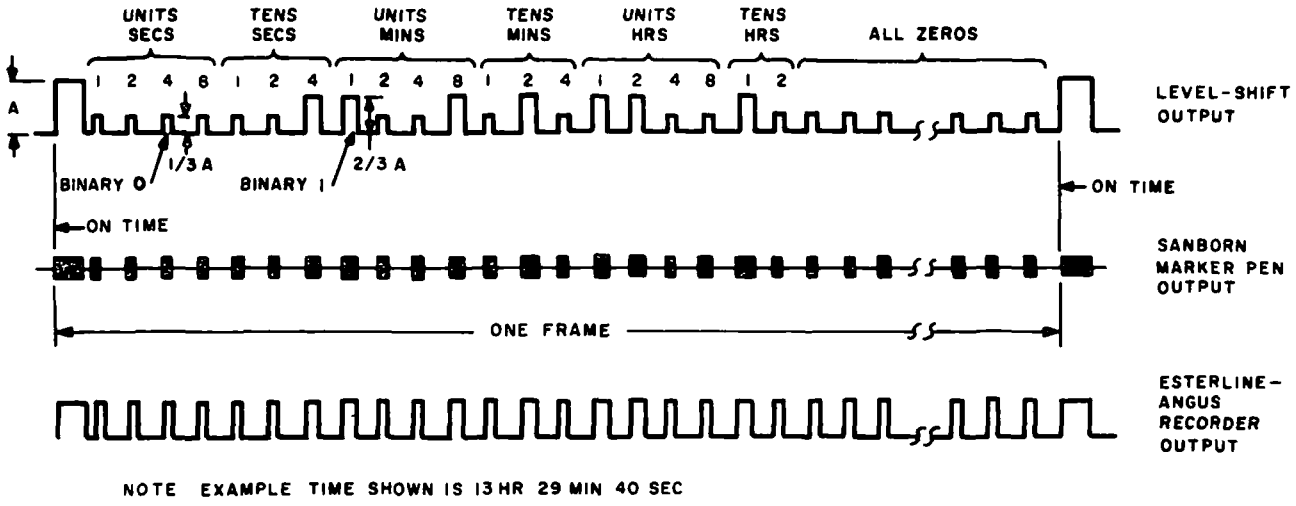


Figure III-4. Digital Time-Marker Equipment, Block Diagram

Two slow-time codes are generated for marking the paper-chart recordings. One of these codes has a 2-pps bit rate and a 30-second frame period and is applied to the marker pens of the two Sanborn telemetry recorders. The other slow-time code has a 0.5-pps bit rate and a 120-second frame rate and is applied to one channel of the Esterline-Angus events recorder. These slow-time-code outputs (as well as other, similar, outputs to the SCAMA output circuits) are buffered by a relay driver and relay unit (EECO Model 861). Each time code consists of a 60-bit serial PDM signal. The pulse widths and sequence are indicated in Figure III-5.

B. PROGRAMMING SUBSYSTEM

A description of the programming subsystem is contained in Volume II, the classified supplement, of this report.



NOTE EXAMPLE TIME SHOWN IS 13 HR 29 MIN 40 SEC

FRAME PERIOD	BIT RATE	REFERENCE MARK	BINARY 0	BINARY 1
30 SEC	2 PPS	400 MS	100 MS	200 MS
120 SEC	0 5 PPS	1600 MS	400 MS	800 MS

Figure III-5. Digital Time-Marker, Slow Time Codes

C. TV RECEIVING EQUIPMENT

1. General

The video data from the ESSA 1 satellite is transmitted to the CDA ground stations over the 235-Mc frequency band. The transmissions are received by separate horizontally polarized and vertically polarized elements of the ground station antenna, and are applied to the TV and IR receiving equipment.

The TV receiving circuit consists of two bandpass filters, two distribution amplifiers, two TV receivers tuned to 235 Mc, and a diversity combiner. Figure III-6 is a block diagram of the TV receiving circuit. The horizontally polarized and vertically polarized outputs from the antenna are applied through bandpass filters before being applied to their respective TV receiver. The bandpass filters, which are used to prevent interference from the command transmitter, are peaked at 236.5 Mc.

The signal output and AGC voltage of both the horizontal and vertical receivers are applied to the same diversity combiner. Similarly, the outputs of the two TV receivers are applied to a second diversity combiner. Each diversity combiner selects the

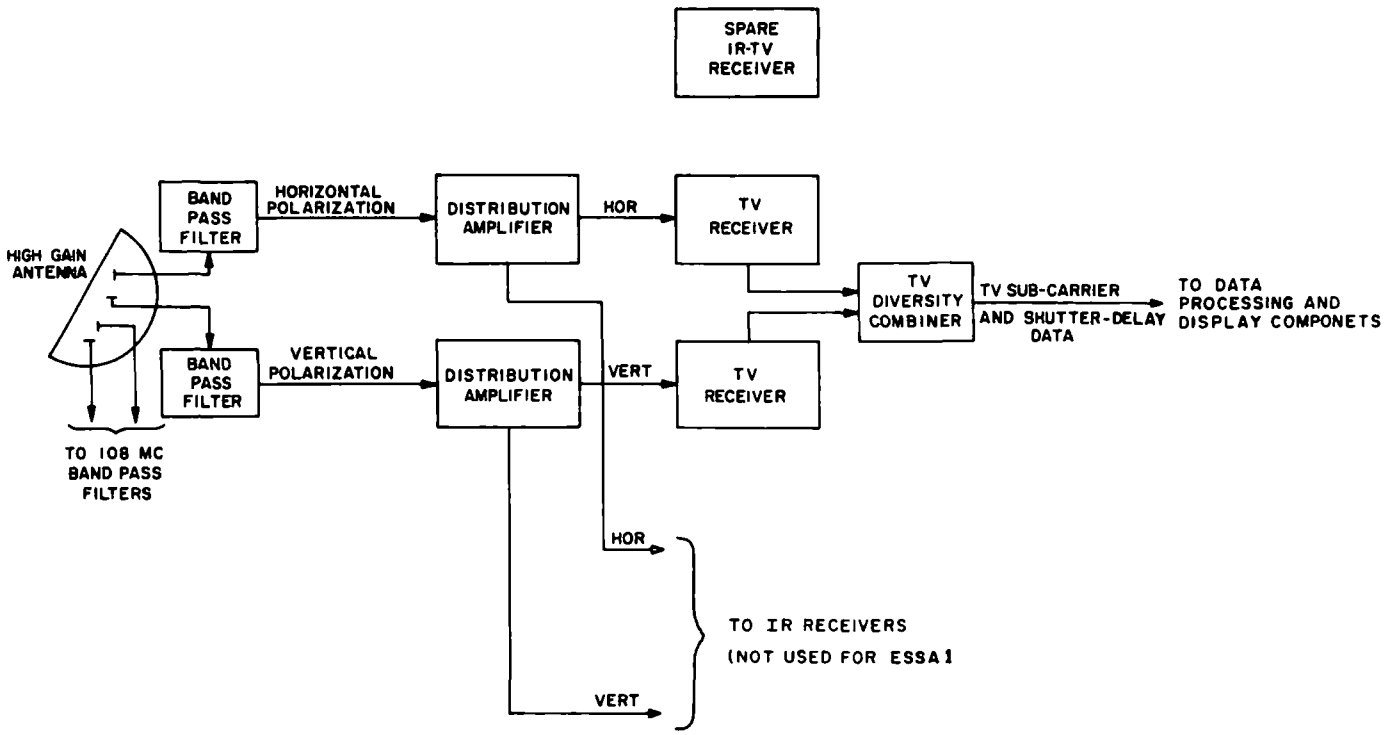


Figure III-6. TV Receiving Equipment, Block Diagram

stronger of the two signal inputs or, if the two are of equal strength, combines the two inputs for application to the succeeding stages of the TV receiving circuits.

The TV receivers used in the CDA ground stations are Nems-Clark Model 1411 or Nems-Clark Model 1412. Complete descriptions and schematic diagrams of these units will be found in their respective instruction books.

2 Diversity Combiners

The diversity combiner unit for the TV receivers is located to the rear of the TV receivers in Rack 11. Each diversity combiner selects the strongest input signal (above the preselected equalization level of AGC) by sensing the levels of the AGC inputs from the two receivers in the same rack, or combines the two signals if the AGC levels are equal. The selected input is amplified and routed to the output for application to an external filter. To minimize distortion, the output of the diversity combiner unit is maintained at 0.1 volt or less.

D. DATA PROCESSING EQUIPMENT

1. General

The data processing equipment provides identification for each TV picture received during a satellite pass by generating a unique index consisting of a sequential frame number that indicates the order in which the picture was received, and source and mode indications which identify the command tone (Direct 1, Direct 2, Playback 1 or Playback 2) being transmitted at the time of picture reception (with a correction for satellite turn-off delay). The frame number is obtained by counting the vertical sync pulses received during the pass, while the source and mode indications are obtained from the ground-station programming equipment. After processing, the index is displayed on the front panel of the kinescope display unit just below the kinescope for photographing. The data is also recorded on the TV tape recorder along with the associated TV picture. The source and mode indications will not necessarily be correct in the "recorders-crossed" mode.

The data-processing equipment also provides for the demodulation of picture-delay data transmitted (on the 10-kc subcarrier) with each TV picture, converting this data into pulses suitable for display, as Z-axis modulation, on the display vidicon.

A block diagram of the data processing equipment, whose major components are a tape and computer control unit and an index generator*, is shown in Figure III-7. The tape and computer control unit combines several signal-processing and control functions, including demodulation and processing of picture-delay data, detection of video subcarrier, and the shaping of command-identifying and readout pulses for the index generator. In addition, the unit incorporates circuitry and front-panel controls for controlling the start, stop, rewind and playback operating functions of the two ground-station tape recorders.

The command indications from the tape and computer control unit are used to set a flip-flop in the index generator which stores the mode (Direct or Playback) and source (Camera 1 or 2) indications. The frame number is stored in a six-bit counter which counts readout pulses from the tape and computer control unit. The mode, source and frame count are transferred into the shift register at the end of the readout pulse.

The outputs of the index generator are 40-kilocycle tone bursts (logical "0"s) and 70-kilocycle tone bursts (logical "1"s) for recording the index data on the ground-station tape recorders. In addition, nine parallel outputs are taken from the nine-bit shift register and applied to two sets of lamp-driver circuits. One set of lamp drivers drives the index indicator on the front panel of the index generator (INDEX NUMBER,

*The Index Generator is itself a sub-section of the sun-angle computer, a unit whose capabilities are fully utilized only during interrogation of the earlier TIROS satellites.

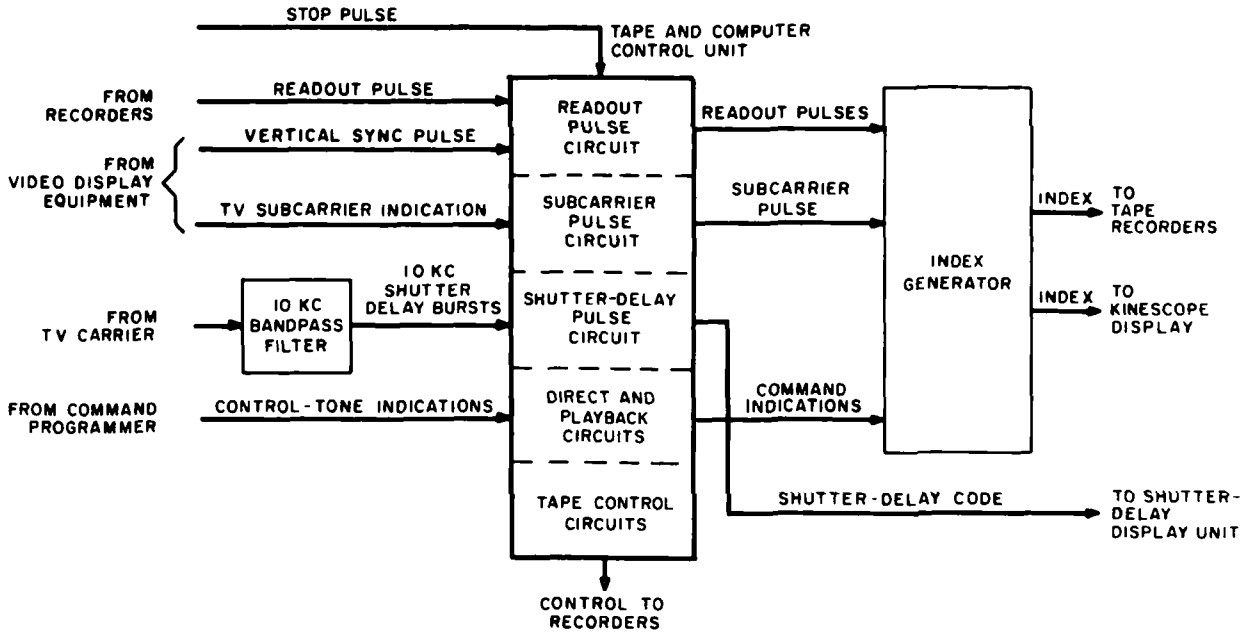


Figure III-7. Data-Processing Equipment, Block Diagram

SOURCE, and TAPE indicators); the other drives identical indicators on the kinescope display unit for photographing with the TV display. For recovery of the recorded source and index data during playback of the magnetic tape recorders, the input to the index generator is connected to the tape recorder output through a tone demodulator which detects the one-millisecond bursts of 40-kc and 70-kc tones from the recorder tapes.

2. Tape and Computer Control Unit

a. General

The tape and computer control unit (located in Rack 5) produces seven outputs, namely: shutter-delay pulses, subcarrier pulses, readout pulses, and delayed Direct 1, Direct 2, Playback 1 and Playback 2 command indications. The unit also controls the operation of the two tape recorders. Functionally, the unit is divided into five main sections: the shutter-delay pulse circuit, the subcarrier pulse circuits, the readout pulse circuit, the direct-and-playback delay circuits, and the tape control circuits. The tape and computer control unit used for the ESSA 1 mission is essentially the same equipment used for previous TIROS missions.

b. Shutter-Delay Pulse Circuit

The shutter-delay pulse circuit which was used to demodulate 10-kc sun pulses for earlier TIROS satellites is used to demodulate 10-kc bursts of shutter-delay code for the ESSA 1 satellite.

The shutter-delay input, consisting of 10-kc bursts which have been recovered from the TV carrier, is heterodyned to a lower frequency (1 kc), detected, and converted to negative-going pulses which are applied to the shutter-delay unit in the TV display equipment.

c. Subcarrier Pulse Circuit

The subcarrier indication from the TV display equipment, a two-second pulse, is converted to negative-going pulses at the output that are applied to the index generator.

d. Read-Out Pulse Circuit

The read-out pulse circuit receives one of two inputs: a stop signal from the command programmer or a read-out gate signal from the sun-angle computer. The signal that is present is converted to a positive-going, 10-microsecond pulse. This readout pulse is applied to the sun-angle computer. The read-out pulse can also be produced by a vertical sync input from the TV-FM demodulator.

e. Direct and Playback Delay Circuits

The direct and playback circuit receives any one of the direct and playback tone indications from the command programmer, delays the input signal for 27 seconds, and then applies the signal to the index generator as a source signal.

f. Tape Control Circuits

Automatic operation of the tape recorders is accomplished using any one of the direct or playback signals from the command programmer. When either a Direct 1, Playback 1 Direct 2 or Playback 2 tone is generated, a RECORD indicator lamp lights and a start signal is sent to the tape recorders.

The tape recorders can be operated manually in each mode by switches S1 (RECORD), S2 (STOP), S3 (REWIND), and S4 (PLAYBACK).

3. Index Generator

During direct television transmission from the satellite to the ground-station, the index generator provides the frame number for each TV picture by counting the read-out pulses (which correspond to the start of each TV frame) applied to the unit from the tape and computer control unit. This is accomplished by a 6-bit counter circuit and a 6-bit shift register. After each readout pulse is counted, the total count in the counter is applied to the shift register and is shifted out, in parallel, to the kinescope display unit, then serially to the ground station tape recorders. Thus, each picture received is assigned a sequential frame number from 1 to a possible 63.

The command-tone indications from the tape and computer control unit are also applied to the index generator to provide an indication of the satellite camera system originating the picture (source), and whether the picture was transmitted directly from the satellite or read out from the satellite tape recorders (mode). Each time that the source or mode changes, the 6-bit counter which generates the frame number is reset. The entire index, therefore, consists of source (1 or 2) and mode (direct or tape) indications, plus a sequential frame number.

Because the ESSA 1 satellite can be operated with its TV camera systems and/or tape recorders cross-coupled (that is, both systems operating in response to a single command tone), the source indications, which are derived from the command tones, do not necessarily represent the satellite camera system originating the pictures. If, for example, the TV camera systems are cross-coupled and either a Direct 1 or Direct 2 command tone is transmitted, the pictures received from the satellite will alternate from camera to camera, but the "source" indication generated by the Index Generator will be dependent on the actual tone transmitted (i.e. 1 for a Direct 1 tone and 2 for a Direct 2 tone). Similarly, if the satellite tape recorders are cross-coupled and a playback tone is transmitted, both recorders will transmit stored pictures from both camera systems to the ground station. The "source" indication for these pictures will also be determined by the actual tone transmitted (i.e. 1 for a Playback 1 tone transmission and 2 for a Playback 2 tone). However, since the pictures from each of the ESSA 1 cameras have a unique, readily identifiable, orientation, no problem in picture identification is presented.

A possible ambiguity in indexing exists when the satellite tape recorders are operated in the cross-coupled mode, since the total number of pictures stored on both recorders may exceed 63 frames, and the 6-bit counter will not be reset by the source or mode indications when the second recorder is read out. In this event, the counter will recycle at the 64th frame, and reassign frame numbers starting with a digital 1. Picture identification is readily resolvable, however, by observation of the real time indication that appears in each frame of the 35-millimeter photographic film.

The index generator makes provisions for playback of a particular frame of a series of TV pictures that have been recorded on the ground-station recorders. The source, mode, and frame number of the desired TV picture is manually set into the index generator, the tape recorder is energized, and the information from the tape is used to generate the readout pulses. The frame numbers of the recorded pictures are then compared with the selected picture, and, when the proper frame appears under the read head of the recorder, the recorder output (the selected picture) is gated to the TV kinescope.

E. TV DISPLAY EQUIPMENT

1. General

The TV display equipment accepts the TV subcarrier received from the satellite, and demodulates, amplifies, and processes the video data for kinescope presentation and film recording. A block diagram of the TV display equipment is shown in Figure III-8.

The input signal to the equipment, a frequency-multiplexed composite of the 85-kc TV subcarrier and the 10-kc picture-delay-code subcarrier, is applied to a filtering network. A 10-kc bandpass filter in the network separates the picture-delay subcarrier from the composite signal, and routes the picture-delay subcarrier to the data-processing equipment for subsequent demodulation and pulse shaping. The 85-kc TV subcarrier is passed by a high-pass filter and applied to the TV-FM demodulator.

The TV-FM demodulator receives and demodulates the FM-modulated TV subcarrier, and provides a stable, low-impedance, video output. In addition, the unit generates vertical sync pulses, corresponding to the leading edge of the subcarrier, which are applied to the vertical deflection circuits in the sawtooth and deflection amplifier, and to the data processing components. The video output from the TV-FM demodulator is applied to the horizontal sync separator, to a monitor oscilloscope, and to the shutter-delay display unit.

The horizontal sync separator generates horizontal sync pulses in phase with the horizontal sync pulses contained in the video signal, and supplies these pulses to the horizontal deflection circuits in the sawtooth and deflection amplifier and to the shutter-delay display unit. The sawtooth and deflection amplifier, in turn, supplies horizontal and vertical deflection voltages and horizontal and vertical blanking pulses to the kinescope in the display and video amplifier.

The shutter-delay display unit interrupts the video signal from the TV-FM demodulator for approximately 100 microseconds each time a horizontal sync pulse is received. During this 100-microsecond period, the unit permits the demodulated picture-delay code from the data processing equipment to be coupled to the input of

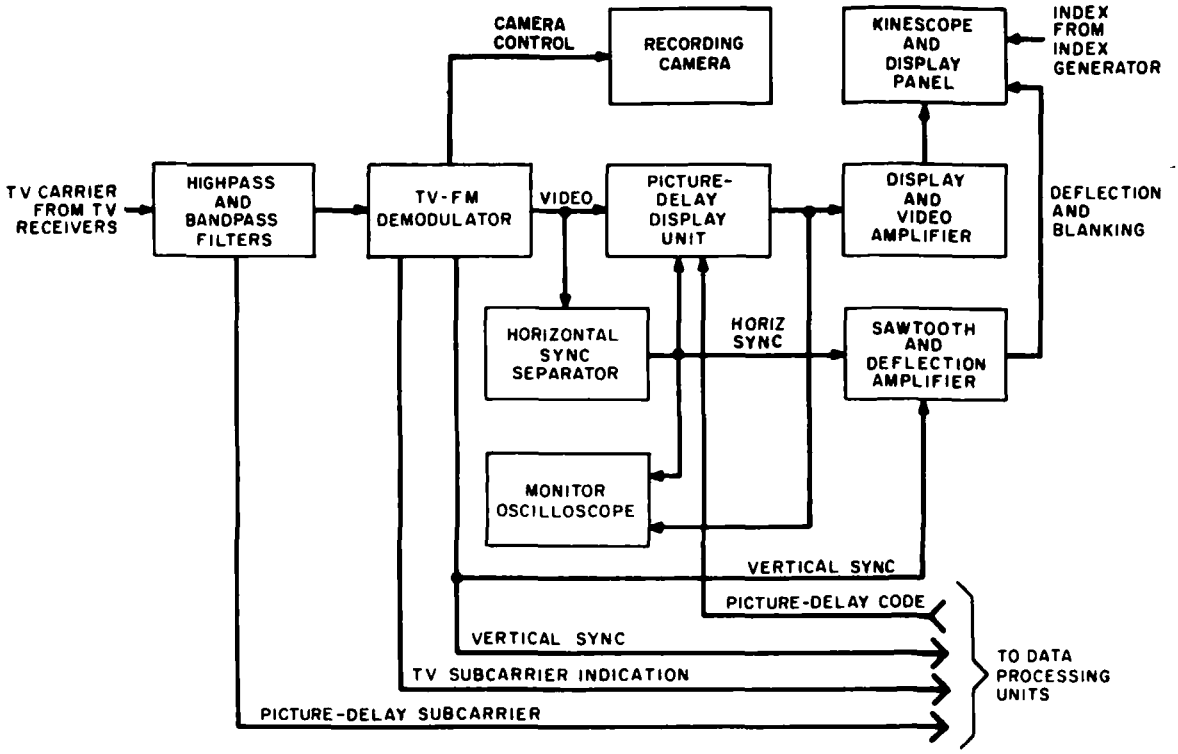


Figure III-8. TV Display Equipment, Block Diagram

the display and video amplifier, and presented on the face of the kinescope. The overall effect of this action is to present the picture-delay code as a black and white bar on the edge of the kinescope display, where it is photographed along with the TV picture.

The display and video amplifier provides the final amplification for the TV video signal and presents the associated TV picture on a display panel. This panel contains indicator lamps that are illuminated by inputs from the data-processing components to provide picture identification data. A camera mount, which permits taking of 35-mm and Polaroid pictures simultaneously, facilitates photographing of the TV display.

2. TV-FM Demodulator

The TV-FM demodulator unit, located in rack number 4, demodulates the television subcarrier input to provide the video signal for the display and video amplifier and horizontal sync separator units. The FM demodulator also generates a vertical sync pulse for the sawtooth and deflection unit, shutter control for the recording camera, and a subcarrier indication for the data processing equipment.

3 Horizontal Sync Separator

The horizontal sync separator unit, located in rack number 4, generates the 250-cps horizontal-sync pulses for the sawtooth and deflection amplifiers, and phases these pulses to coincide with the sync pulses in the received video signal.

4. Shutter-Delay Display Unit

The shutter-delay code (also referred to as picture-time-delay code), which signifies the shutter time of individual video frames, is transmitted from the ESSA 1 satellite as a keyboard modulation of a 10-kc subcarrier transmitted with the 85-kc video subcarrier by the TV transmitter. The demodulation and detection channel in the tape and computer control unit used for this 10-kc subcarrier is the same channel that was used for the sun-angle code on previous TIROS satellites. The code is recorded with the associated video frame on the kinescope and on one pen of the Esterline-Angus events recorder.

The shutter-delay display unit (which is physically located in the horizontal sync separator) interrupts the video from the TV-FM demodulator during the first 100 microseconds of each horizontal TV line, and couples the picture-delay code to the video and display amplifier during this period. This causes the pulse-code-modulated (PCM) picture-delay code to be presented as a narrow black-and-white band on the edge of the kinescope display.

This band along the left-hand edge of the TV picture can be decoded to obtain the actual delay code when the TV picture is interpreted. By suitable adjustment of the period of the one-shot multivibrator, the width of the strip can be made sufficiently narrow so as not to subtract appreciably from the picture area. A pulse-width control provides for adjustment of the one-shot multivibrator period over a range from zero to 120 microseconds.

In addition to the presentation on the display kinescope, the picture-delay code is also recorded on the ground-station events recorder. Whenever a logical "1" is present in the code, a +28 volt level is coupled to the events recorder, and the logical "1" is displayed as a deflection of the recorder pen.

5 Sawtooth and Deflection Amplifier

The sawtooth and deflection amplifier, located in rack number 4, generates the horizontal and vertical deflection voltages and the horizontal and vertical retrace blanking pulses for the display kinescope. The horizontal and vertical sawtooth and deflection circuits are very similar in design, differing mainly in the repetition rate, slope, and duration of the output waveforms they produce. Each generates a sawtooth voltage

waveform, which is applied by a push-pull amplifier to its respective set of deflection plates in the display kinescope, and a rectangular blanking pulse which is applied to the screen grid of the display kinescope during the retrace interval. Each circuit is provided with slope controls (for both trace and retrace slopes) and with centering, sawtooth amplitude, and push-pull amplifier balance controls. The vertical sweep can be disabled for making focus and brightness adjustments of the kinescope.

The retrace blanking circuit receives input pulses from the horizontal and vertical deflection circuits, combines the input pulses, and applies them to the display and video amplifier unit as horizontal and vertical retrace blanking pulses.

6. Display and Video Amplifier

The display and video amplifier, located in rack number 4, amplifies the demodulated video signals and presents them on a kinescope for photographic recording. The front panel of the unit also incorporates the indicator lamps which are energized by the data processing components and which display ancillary data pertaining to picture identification.

The video signal is applied to the kinescope through a relay. When the relay is in the de-energized position, the image on the kinescope is negative; when the relay is in the energized position the image on the kinescope is positive. (The photographic transparencies made from these displays are the opposite "polarity" from the kinescope images.) The brightness of the kinescope picture is controlled by either the POS control (when a negative image is displayed on the kinescope) or the NEG control (when a positive image is displayed on the kinescope) which adjusts the bias on the kinescope.

The vertical and horizontal retrace blanking pulses from the sawtooth and deflection amplifier are amplified and clamped before being applied to the screen grid of the kinescope. These pulses cut off the kinescope during horizontal and vertical retrace. The front panel of the display and video amplifier accommodates indicator lamps which indicate the index for each TV picture. These indicators are energized by signals from the data processing equipment.

The recording camera is mounted on the front of the display and video amplifier unit, and mounted such that its field-of-view takes in the kinescope presentation and the index indicators. The camera shutter is tripped automatically by pulses from the FM demodulation unit each time a picture is received, and film advance after each picture is automatic. A control for manual camera operation is also provided.

7. Monitor Control Unit

The monitor control unit, located in rack number 5, provides switches and relays for selecting the source for the data display components (i.e., either the TV receivers during a satellite pass, or the ground station tape recorders for tape readout), determining whether the kinescope presentation will be positive or negative, and controlling the d-c restorer and demodulator circuits in the FM demodulator unit. The unit also contains the ON-OFF switches for the 32-volt power supply, the three 300-volt power supplies, the high-voltage power supply, and the index generator power supply.

F. BEACON-DATA RECEIVING AND RECORDING SUBSYSTEM

1. General

For the ESSA 1 mission, the beacon-data receiving and recording subsystem includes recording equipment, along with signal switching arrangements, to provide for recording the two subcarrier channels of each beacon. A telemetry calibrator unit allows checking of the entire subcarrier demodulation and recording process. The two beacon frequencies, both of which operate in the 136-MC band, are each amplitude modulated by two subcarriers, one at 1300 cps, the other at 2300 cps. The 1300 cps subcarrier on beacon No. 2 and the 2300-cps subcarrier on beacon No. 1 are continuously frequency-modulated, in turn, by satellite attitude data. The remaining two subcarriers (2300 cps on beacon No. 2, 1300 cps on beacon No. 1) are frequency modulated on a time-shared basis by the attitude data, by two housekeeping telemetry channels, and by telemetered signals from the camera-trigger, solar-aspect-indicator, MBC, and digital-command-verification circuits. The timing and the allocation of this data are detailed in Part II, Section D, of this document.

The beacon data receiving and recording subsystems also provide for the selective restoration of this data from the subcarriers, and for proper routing of the restored data to either of two 2-channel Sanborn paper-chart recorders, one of which is assigned to attitude data, the other to telemetry data.

2 Functional Description

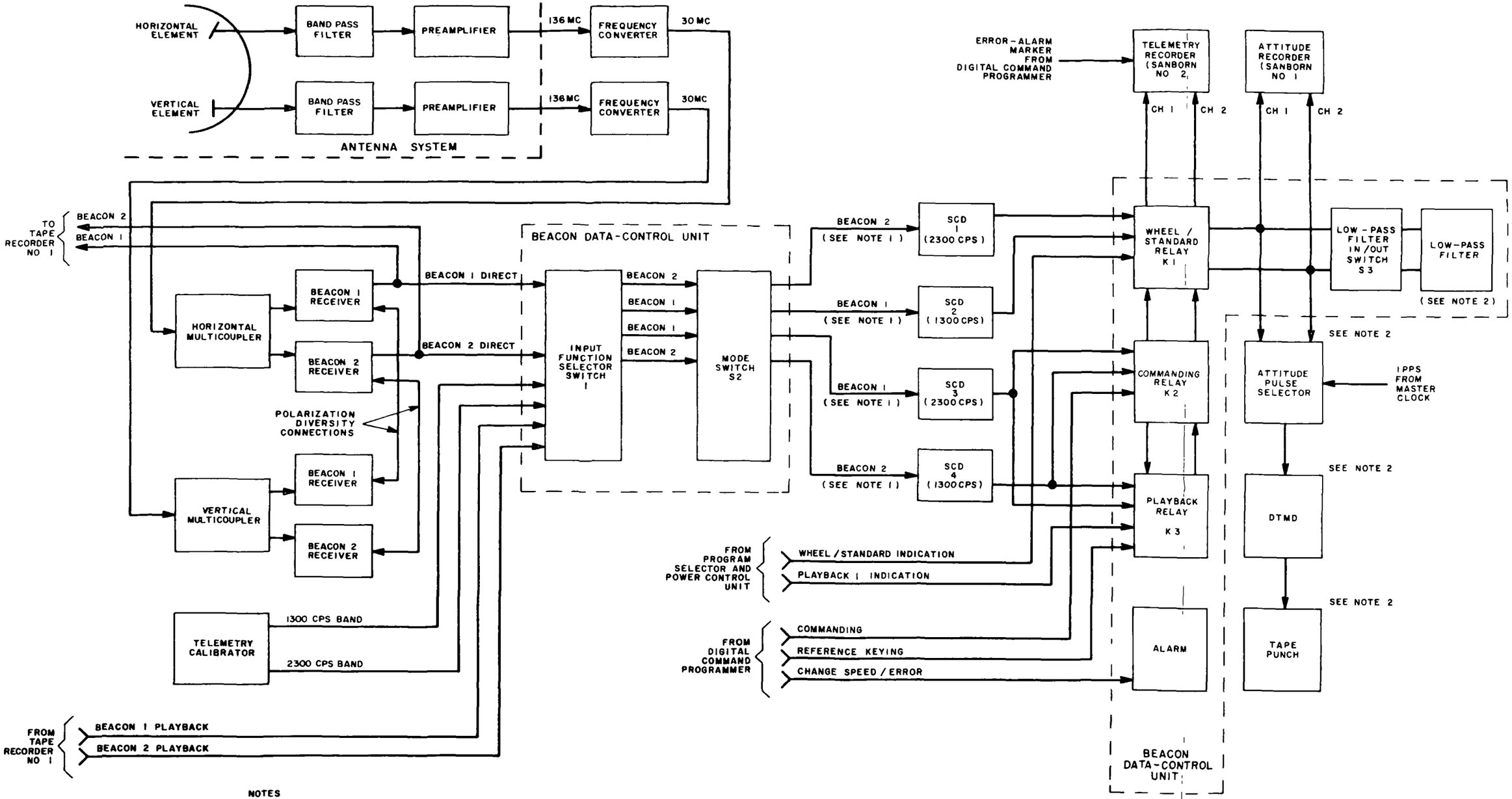
A block diagram of the beacon-data receiving and recording equipment is shown in Figure III-9. The horizontal and vertical components of the two beacon carriers are received by separate vertically and horizontally polarized antenna elements, and are applied through 136-Mc bandpass filters to separate preamplifier stages located at the antenna. The amplified RF signals are brought from the antenna to a pair of 136-to-30-Mc, "Tapetone", crystal-stabilized frequency converters, and the 30-Mc (approximately) output from each of the converters is applied to a pair of Signal Corps type R390-A receivers.

To provide for the reception of both beacon frequencies (136.23 Mc from beacon No. 2 and 136.92 Mc from beacon No. 1), one of the receivers in each pair is tuned to 30.23 Mc, while the other is tuned to 30.92 Mc. Each pair of receivers of the same frequency but of opposite polarization is cross-connected, providing for polarization diversity operation and producing an optimum audio signal at the common output of the pair.

The audio outputs from the receivers, each of which is a composite of the 1300-cps and 2300-cps beacon subcarriers, are applied to separate channels of tape recorder No. 1 in the ground-station recording equipment, for permanent recording on magnetic tape. The outputs are also applied to the INPUT FUNCTION SELECTOR switch, S1, in the beacon data-control unit. Switch S1, when placed to the RECEIVERS position, accepts the composite audio signal from the receivers and supplies four outputs - two of the beacon-1 composite audio and two of the beacon-2 composite audio - to the MODE switch, S2. When prerecorded TV pictures from the ground-station tape recorders are to be played back, INPUT FUNCTION SELECTOR switch S1 is placed to the TAPE REC 1 position, and the previously recorded attitude data associated with the played back pictures is similarly coupled to the MODE switch. The third position of INPUT FUNCTION switch S1, FREQ CAL, connects the outputs from the telemetry calibrator unit to the MODE switch (thence to the remainder of the subsystem). In this position, the accurate audio frequencies generated in the telemetry calibrator are made available for subsystem calibration.

The MODE selector switch, S1, provides the proper switching path for routing the composite audio signals from the beacons to the proper subcarrier discriminators (SCD's), and therefore, to the proper Sanborn recorder channels. In normal operation, this switch will always be placed in the BEACON NO. 1 & 2 position, and the signals will be coupled to the SCD's as shown in Figure III-9, with the composite audio from beacon No. 2 applied to SCD 1 and SCD 4, and the composite audio from beacon No. 1 applied to SCD 2 and SCD 3. However, if one of the satellites beacon transmitters should fail, the MODE switch would be placed to the switch position (BEACON NO. 1 or BEACON NO. 2) corresponding to the remaining operative beacon, in order to provide both attitude and telemetry data to the Sanborn recorders.

The four composite-audio signals, two containing the beacon No. 1 subcarriers and two containing the beacon No. 2 subcarriers, are brought out from the beacon data control unit to the four SCD's. Each of the SCD's is, essentially a combination of a bandpass filter, a discriminator, and a low pass filter which acts to separate a particular subcarrier-frequency band from the composite audio signal at its input, and to demodulate the subcarrier, restoring the d-c level variations which were used originally to modulate the subcarrier in the satellite. The composite audio from beacon No. 2 is applied to SCD-1, a 2300-cps discriminator unit, and to SCD-4, a 1300-cps unit. The signals at the output terminals of the units, therefore, are restorations of the modulation on the 2300-cps and 1300-cps subcarriers, respectively. Similarly, the outputs from the SCD-2 and SCD-3 are a restoration of, respectively, the 1300-cps



NOTES

1 INPUTS TO SCD'S SHOWN FOR POSITION 1-2 OF MODE SWITCH S2 ON BEACON DATA-CONTROL UNIT (NORMAL OPERATION)

2 UNITS USED IN STANDARD TIROS OPERATION ONLY

Figure III-9. Beacon-Data Receiving and Recording Subsystem, Block Diagram

and 2300-cps subcarrier modulating signals present in the beacon-1 transmission. The outputs from the SCD's are returned to the beacon data control unit and applied to contacts of three control relays: K1, K2, and K3.

The signal from SCD-1, which contains the attitude data from 2300-cps subcarrier of beacon No. 2, and the signal from SCD-2, which contains the attitude data from the 1300-cps subcarrier of beacon No. 1, are applied through normally closed contacts of "Wheel/standard relay" K1 of the beacon data control unit to the two channels of the attitude recorder (Sanborn recorder No. 1). (Relay K1 is de-energized in the Wheel operating mode.) Thus, as soon as the satellite is acquired, the attitude data starts to be recorded, and is recorded continuously throughout the duration of the satellite pass.

The data from the two telemetry-carrying subcarriers, the 1300-cps carrier on beacon No. 2 and the 2300-cps carrier on beacon No. 1, are taken from SCD-4 and SCD-3, respectively, and applied to contacts on "commanding" relay K2 and playback relay K3. During the direct-camera sequences, and during the portions of the playback sequence when the FSK digital commands are not being transmitted. These signals are applied through normally closed contact pairs on K2 and K1, to the two channels of the telemetry transmissions are recorded on the paper chart.

When the FSK command tones are transmitted to the satellite during the Playback Sequence, the satellite's telemetry system begins transmitting command-verification signals to the ground station on one of the beacon frequencies. If the FSK transmission is programmed to occur during a Playback 1 tone transmission, this data will be modulated onto the 2300-cps subcarrier of beacon No. 1; if the FSK transmission occurs during Playback 2 tone transmission, the data will be modulated onto the 1300-cps subcarrier of beacon 2. Also, during FSK tone transmission, the digital command programmer generates a "reference-keying" signal which contains d-c levels analogous to the bit content (i.e., the logical "1's" and logical "0"s) of the FSK command word as it is being transmitted. Both of these signals are presented side-by-side on the telemetry recorder during each 3.5 second period of FSK command-word transmission, with proper switchover being automatically accomplished by the beacon data control unit.

Figure III-10 is a simplified schematic diagram of the command-verification switchover circuit in the beacon data control unit. During those portions of the command program when FSK tones are not being transmitted, the outputs from SCD-3 and SCD-4 are applied to the telemetry recorder channels through normally closed contacts of K2. However, during each 3.5-second FSK-tone transmission, a +28-volt level from the digital command programmer is applied to the solenoid of K2, energizing it, and causing the telemetry recorder channels to be fed through playback relay K3.

If the FSK command word is being transmitted during a Playback 2 tone transmission, relay K3 will be de-energized, and the signal paths will be as shown in Figure III-10,

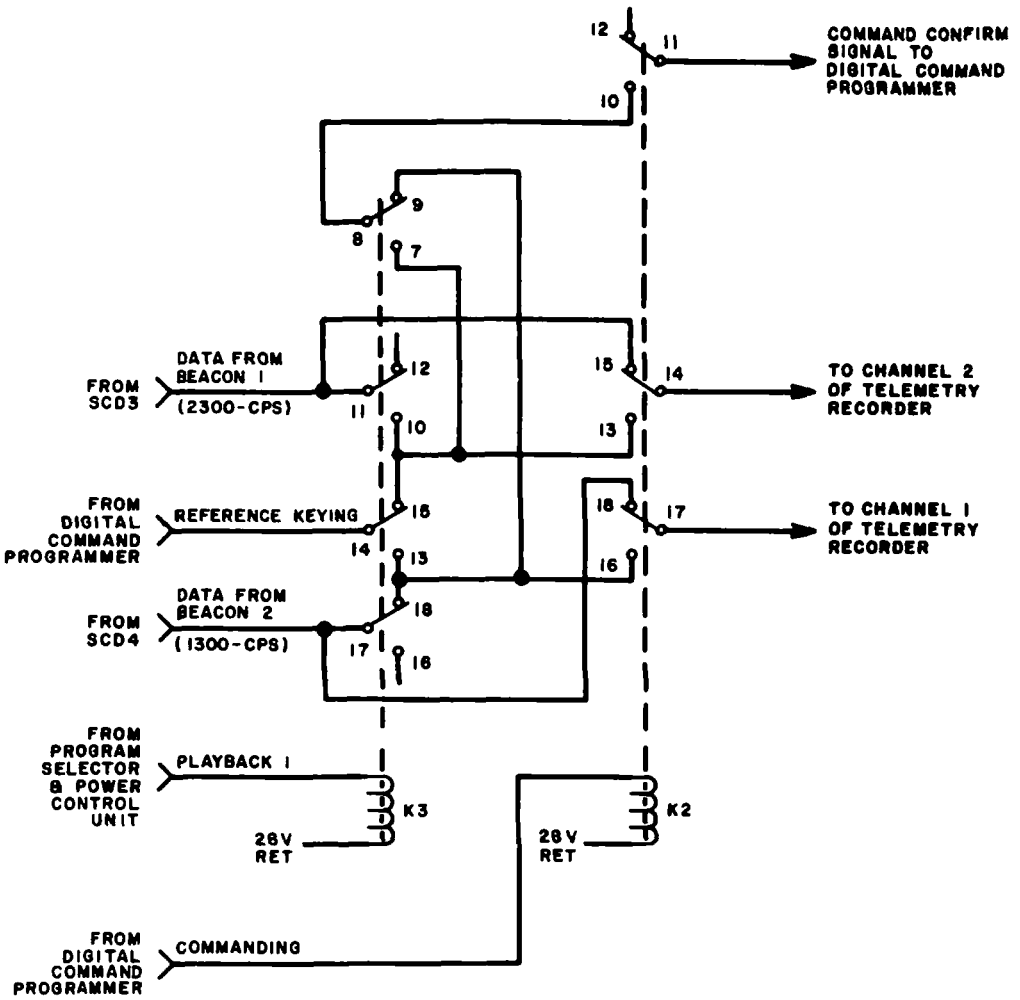


Figure III-10. Command Verification Switchover Circuit, Simplified Schematic Diagram

with the command-verification signal from the 1300-cps subcarrier of beacon No. 2 applied through K3 (17-18) and K2 (16-17) to channel 1 of the telemetry recorder, and the "reference-keying" signal from the digital command programmer applied through K3 (14-15) and K2 (13-14) to channel 2. However, if the FSK transmission is programmed to occur during Playback 1 tone transmission, a level from the program selector and power control unit (in the programming section of the ground section) will energize K3. The command-verification signals derived from the 2300-cps subcarrier of beacon No. 1 will then be applied to telemetry recorder channel 2 through K3 (11-10) and K2 (13-14), while the "reference-keying" signal will be coupled through K3 (13-14) and K2 (16-17) to channel 1.

It can be noted from Figure III-10, that no matter what the condition of K3, the output at contact 8 of that relay will always contain the command-verification data transmitted from the satellite. This data is coupled to the digital command programmer through K2 (10-11) during FSK tone transmission and compared, on a bit-by-bit basis, to the signal being generated. If a discrepancy should exist between the generated and received signal, an energizing voltage would be applied to an audible alarm in the beacon data control unit, alerting operating personnel, and the bit containing the error could be isolated by reference to the strip chart on the telemetry recorder. (The alarm circuit is also energized at the beginning of the initial FSK command-word transmission to alert the operator to increase the chart speed of the telemetry recorder during command verification.)

In addition to the recording of the attitude data on Sanborn recorder No. 1, a readout facility, which refines the raw attitude data from standard TIROS satellites into standard 5-hole BAUDOT (teletype) code characters, and presents this data in a punched-tape format, is provided. This facility comprises an attitude pulse selector, a digital time measuring device (DTMD) and a paper-tape punch.

3. Telemetry Calibrator

The telemetry calibrator consists of two sets of eleven audio oscillator circuits, one set that generates frequencies (at 20-cps intervals) over the 1200-cps to 1400-cps frequency-deviation range of the 1300-cps subcarrier, and another that generates frequencies (at 35-cps intervals) over the 2125-cps to 2475-cps frequency deviation range of the 2300-cps subcarrier. The output from each oscillator is brought out to contacts on a 14-position, 2-deck, wafer switch; the oscillators associated with the 1300-cps subcarrier are supplied to one deck and those associated with the 2300-cps subcarrier to the other. As the switch is rotated through its positions, the entire deviation range for both subcarriers is sampled. The first two positions of each deck of the switch are connected to a front-panel coaxial-type connector to permit coupling of external calibration frequencies to the subsystem.

The outputs from the wipers on each deck of the switch are applied to power amplifier stages which provide power amplification for the audio signals. A 2-line output from each power amplifier is applied to the INPUT FUNCTION SELECTOR switch in the beacon data control unit for normal subsystem calibration. In addition, the outputs from each frequency band are available for test purposes. The outputs from the power amplifiers are also combined in a resistor network, and the resultant composite audio signal, which simulates the output of the beacon receivers is made available at a front-panel coaxial connector.

G. TV TAPE-RECORDING EQUIPMENT

1. General

The TV information received from the satellite, along with related identification and other auxiliary data, is recorded on two standard Ampex, Series FR-100C of FR-100A, seven-channel magnetic tape records. The tape recorders are started automatically at the beginning of each ground-to-satellite contact by a signal from the ground-station programming equipment, and are operated throughout the pass at a recording speed of 60 inches per second.

Table III-2 lists the channel assignments for the tape recording equipment. The Ampex series FR-100A is used at Wallops Island, while the Ampex series FR-100C is used at the Fairbanks station.

The Ampex tape recorders perform a similar function during the interrogation of standard TIROS satellites, but the channel assignments and the data recorded differ. In addition, an IR quick-look facility associated with tape recorder No. 2 provides for evaluation of IR data transmitted from the standard TIROS satellites. This capability has no function in ESSA 1 operations.

H. EVENTS RECORDER

1. General

A 20-channel Esterline-Angus events recorder, Model AW (mounted in rack number 18 Wallops Island and Fairbanks station) is used to provide a direct, real-time record of the command program for use in checking equipment and operator performance prior to a pass. The recorder also provides a permanent record of the commands sent to the satellite during an actual pass.

The events recorder is capable of both manual and automatic operation. Manual operation is used during station troubleshooting and maintenance procedures, while automatic operation is the normal operating mode of the equipment.

2. Functional Description

The events recorder monitors commands sent to the satellite from both Program A and Program B in the ground station command and control equipment, and also monitors the TV receivers (from their AGC circuits) to determine when their respective inputs rise above a certain value, thereby indicating the start of data reception. Other events monitored by the unit include operation of the kinescope-camera shutter and other miscellaneous commands and addresses sent to the satellite from the ground station.

TABLE III-2. AMPLEX RECORDER ASSIGNMENTS AT THE WALLOPS ISLAND AND FAIRBANKS CDA STATIONS

Channel	Signal Recorded	Record and Playback Mode	Playback Speed (in per sec)
	RECORDER NO. (RACK 9)		
1	TV Subcarrier from High-Pass Filter	Direct	60
2	Index from Index Generator	Direct	60
3	10-kc Subcarrier*	Direct	60
4	Time Code	Frequency Modulated	60
5	Beacon No. 2 Subcarriers	Frequency Modulated	60
6	Program Events	Direct	60
7	Beacon No. 1 Subcarriers	Frequency Modulated	60
	RECORDER NO. 2 (RACK 10)		
1	TV Subcarrier from High-Pass Filter	Direct	60
2	Index from Index Generator	Direct	60
3	10-kc Subcarrier*	Direct	60
4	Composite IR**	Frequency Modulated	Not Used
5	IR Events (Channel 4, Quick-Look Demodulator)**	Frequency Modulated	3-3/4
6	IR Clock (16.5 kc, Quick-Look Demodulator)**	Direct	3-3/4
7	10-Kc	Frequency Modulated	Not Used

*The 10-kc subcarrier is modulated with shutter-delay code data in ESSA 1 operations, with raw sun-angle data in the earlier TIROS missions.
**Not present in ESSA 1 operations.

An ON-OFF pushbutton supplies power to the recorder and an AUTO-MAN pushbutton permits selection of automatic or manual operation. In automatic operation, the recorder chart is set to a convenient 1-minute interval mark, and, when the first alarm of the program sequence is generated by the master clock, a start signal is applied to the recorder which energizes the recorder motor and sets the chart in motion. Real-time start of the chart is, therefore, the alarm A1 time of the particular satellite pass. In manual operation, the recorder motor is energized continuously, and real time must be computed from the timing pulses which are presented on channel 8 of the recorder. The recorder operates at a speed of 6 inches per minute.

Table III-3 presents a list of the channel assignments for the ESSA 1 events recorder. The channel assignments for ESSA 1 differ from those for the standard (axial-mode) TIROS satellites, and switching of the inputs to the differing channels for ESSA 1 operation is accomplished by a bank of relays which are energized by levels from the program selector whenever the ground station is set up to program ESSA 1.

I CDA-SCAMA INTERFACE EQUIPMENT

The CDA-SCAMA interface equipment at the CDA stations allows transmission of the beacon-subcarrier data and program events to TTCC via the telephone-tape SCAMA circuits. Test signals, including a series of tuning-fork-controlled tones from the telemetry calibrator unit can also be placed on the SCAMA lines for calibration or checking purposes. A functional block diagram of the CDA-SCAMA interface equipment is shown in Figure III-11.

Switching and selection of data to be transmitted is accomplished by means of the CDA data output unit (CDOU). The CDOU performs the following functions:

(1) Selects either direct beacon data from the beacon receivers, taped beacon data from Ampex magnetic tape recorder No. 1, or calibration signals from the telemetry calibration unit; the selection is handled by means of an INPUT switch.

(2) Controls selection of beacon subcarriers by means of MODE switch. Four bandpass filters, one for each beacon-subcarrier channel, allow selection of one of the two subcarriers (1300 cps or 2300 cps) on each beacon signal. Resistive networks provide proper impedance matching for the bandpass filters, and allow the outputs of a pair of filters to be summed.

(3) Allows recording and reproduction of the multiplexed and FSK-modulated programmer events data signals by matching the input and output impedance of channel 6 of magnetic tape recorder No. 1 through matching resistors and transformers.

TABLE III-3 ESSA 1. EVENTS RECORDER CHANNEL ASSIGNMENTS

Channel	Operation
1	Direct 1 Commands
2	Direct 2 Commands
3	Playback 1 Commands
4	Playback 2 Commands
5	Auxiliary 1 Commands
6	Auxiliary 2 Commands
7	Auxiliary 3 Commands*
8	Timing Markers
9	Program B
10	Address and Spin-up Commands
11	Digital Commands
12	Timing Markers
13	Timing Markers
14	Attitude Control Commands
15	Picture-Delay Data
16	Monitor Shutter Indication
17	TV Receiver 1
18	TV Receiver 2
19	Not Used
20	Not Used

*Not used on ESSA 1

Since the beacon subcarrier signals are attenuated by the bandpass filters and matching-impedance networks, it is necessary to amplify the combined filter outputs to attain a 0-dbm level for feeding an input to the SCAMA circuits. For this purpose, the unused audio amplifier section on each of the two R390 beacon receivers is connected as a buffer amplifier between the output of one pair of filters and the input to the SCAMA circuits. These amplifiers also provide a gain control for setting proper output level and a meter for monitoring that output level.

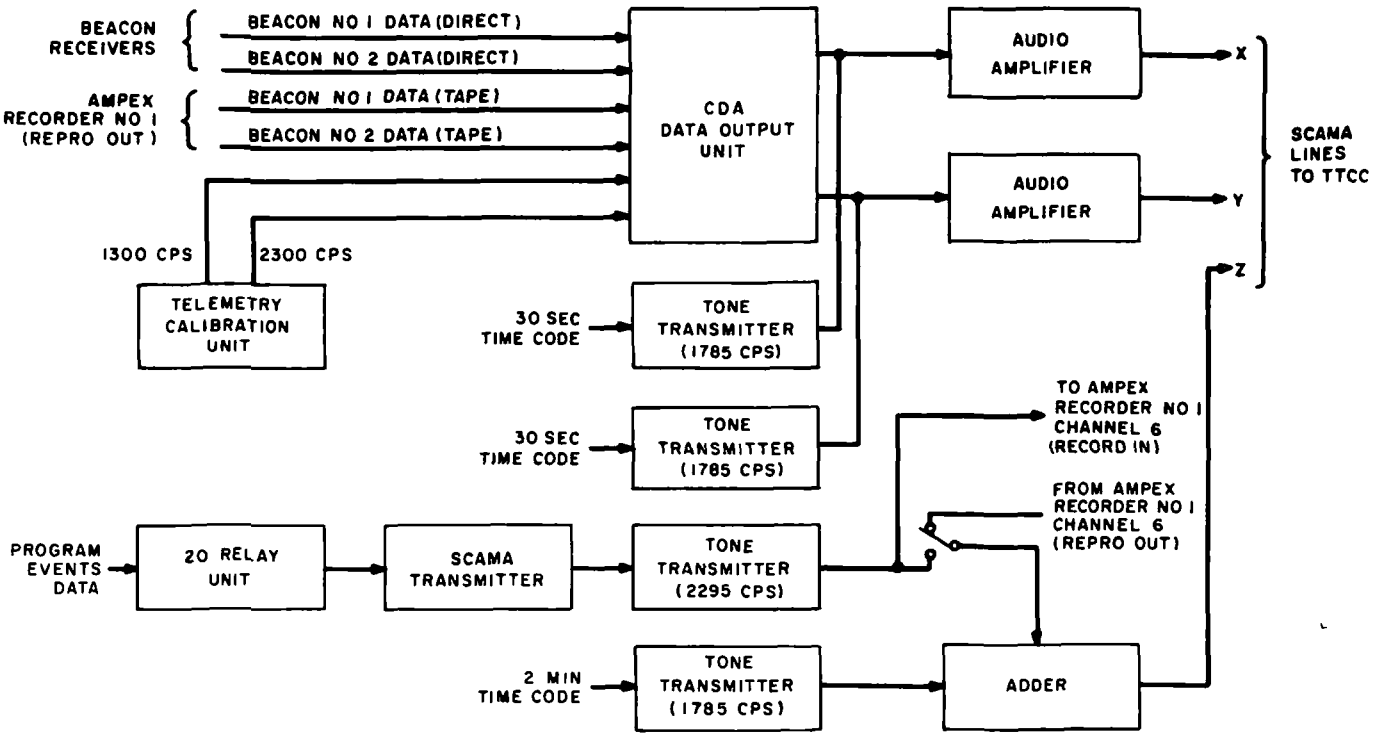


Figure III-11. CDA-SCAMA Interface Equipment, Block Diagram

Program events data signals are taken from the Esterline-Angus events recorder and applied to a bank of 20 relays which, in turn, provide contact closures to a scanner-transmitter unit, Model QST-24. The QST scanner transmitter, QT-30 tone transmitters, and associated equipment are standard, government-furnished, commercial units built by Quindor Electronics Inc. The QST-24 unit multiplexes the relay closure data into a signal which is FSK-modulated onto a 2295-cps subcarrier by a QT-30 tone transmitter for transmission over the SCAMA circuit.

Two other QT-30 tone transmitters FSK-modulate the 1785-cps subcarriers with a time code (2-pps bit rate). These signals are combined with the beacon signals at the outputs of the band-pass filters of the CDOU.

The frequency shifter unit (FSU) shown in Figure III-11 was added for the TIROS X program to allow the beacon No. 2 subcarrier to be shifted from 1300* to 2300 cps, so that this signal could be mixed in the CDOU with the 1300-cps beacon No. 1 subcarrier for transmission over the SCAMA lines to TTCC.

*On TIROS X, the satellite-borne telemetry and tracking subsystem included only one subcarrier oscillator for each beacon transmitter, with each subcarrier oscillator operating at 1300 cps.

The FSU comprises a 2300-cps voltage-controlled oscillator (VCO), a power supply, and associated circuitry. Switching facilities are also provided to allow the FSU to be switched from TIROS X operations (SHIFT mode) to ESSA 1 operations (NORMAL mode).

In the NORMAL mode of operation, after the beacon No. 2 subcarrier is applied to the FSU it is immediately routed out again, to the CDOU, bypassing the VCO.

J. POWER SUPPLIES

1. General

The power supplies used in the CDA ground stations are listed in Table III-4.

2. Relay Power Supply (Rack 2)

The relay power supply, located in rack 2, supplies 24-volt power for energizing control relays in units housed in racks 1, 2, and 3. The unit also contains four relays, two that control the 115-volt a-c input for the 300-volt power supply in rack 1, and two that control the 115-volt a-c input for the 300-volt power supply in rack 3. Each line is fused and provided with a fuse-indicator lamp. The 24-volt power is obtained from a transformer, full-wave bridge rectifier and filter. The 24-volt d-c supply is provided with an ammeter and voltmeter in the output circuit.

3. Relay Power Supply (Rack 5)

The relay power supply located in rack 5 supplies 24 volts d-c to energize control relays in the video control and power equipment. It also contains three control relays for the power supplies in racks 4 and 5. Each line is fused and provided with a fuse-indicator lamp. The 24-volt power is obtained from a transformer, full-wave bridge rectifier, and filter. The output circuit is provided with an ammeter and voltmeter.

K. GROUND STATION ANTENNAS

1. Wallops Island

Two receiving antennas are used at the Wallops Island station; one for the 235-megacycle band, the other for the 136-megacycle band. The 235-megacycle antenna, a General Bronze multi-element array with a 40-foot diameter ground plane, has a gain of 29 db. This antenna has an automatic tracking capability.

TABLE III-4. POWER SUPPLIES AT THE CDA GROUND STATION

Location	Use	Manufacturer and Model
Racks 1 and 3	300 volts dc	Lambda C481M
Rack 2	300 volts dc	Lambda C481M
Rack 2	28-volt relay supply	RCA (Dwg. No. 1177178)
Rack 4	high voltage	Neutronic 22 MR-VS-MOD
Rack 5	300 volts dc (3 ea.)	Lambda C481M
Rack 5	32 volts dc	Electronic Research Assoc. TR 36-4M
Rack 6	sun-angle computer (index generator) supply (not used on ESSA 1)	—
Rack 8	12 volts dc	Lambda LA-100-03BM
Rack 8	6 volts dc	Lambda LT2095AM
Rack 11	300 volts dc	Lambda C481M
Racks 14 and 15	command transmitter supplies	—
Rack 16 (PMR)	300 volts dc	Lambda C481M
Rack 20	quick-look demodulator supply	—
Rack 32	28 volts dc	Lambda LA100-03BM
Rack 34	28 volts dc	Lambda LA100-03BM
Rack 34	300 volts dc	Lambda C481M

The 136-megacycle antenna, a Kennedy Radiquad, has a gain of 18 db and incorporates a monopulse type of tracking system for automatic tracking. This antenna is mounted on a quadripod structure which is located approximately 300 feet from the 235-megacycle antenna. A small building beneath the antenna houses the command transmitters and a portion of the antenna servo equipment.

Either antenna can be slaved to the other. Initial acquisition of the satellite is obtained by tracking on the 136-megacycle beacon. During this time, the 235-megacycle antenna is slaved to the 136-megacycle antenna. Conversely, after the satellite's TV transmitter comes on, tracking is switched to the 235-megacycle antenna, and the 136-megacycle antenna is slaved to it. If the 235-megacycle signal should fade while the 235-megacycle antenna is tracking, tracking can immediately be switched to the 136-megacycle antenna. In the event of a tracking failure affecting both antennas, control

can be switched to the antenna programmer in the command and control equipment or to a manual tracking mode which is also available.

The command transmitter antenna is a circular-polarized Yagi array with 12-db gain in the 150-megacycle band. This Yagi array is located at the center of the 136-megacycle array.

2. Fairbanks

The receiving antenna at the Fairbanks station is a Blaw-Knox parabolic type, 85 feet in diameter, equipped with an x-y two-axis mount. Gain in the 235-mc band is 33 db. The antenna has auto-tracking capability in the 136-megacycle band in the monopulse mode; however, no auto-tracking capability is present at the TV frequency.

L. AUXILIARY EQUIPMENT

1. Calibrator

The calibrator, located in rack 12, is a test and calibration unit that is used for pre-pass operational checks of the CDA ground stations, and for testing, calibration, and troubleshooting of television-receiving system, and the shutter-delay display circuits. The calibrator is composed of three major functional areas: a video test pattern generator, an 85-kc subcarrier oscillator, and a sun-angle simulator.

The video test pattern generator generates a 500-line pattern having equi-distant vertical and horizontal lines that can be used for checking kinescope display size, aspect ratio, sweep linearity and focus. Each line of video in the pattern is shaded in fourteen graduated steps from black to white for checking brightness level and density linearity.

The 85-kc subcarrier oscillator is provided to check the operation of the video monitor system. The 85-kc subcarrier is modulated by the composite video signal generated by the test pattern generator section (or any other video signal) and can be automatically gated to a single frame of video once every two seconds, thereby simulating actual TV system operation.

The sun-angle simulator circuits generate a series of short, medium, or long 10-kc bursts which simulate picture-data code data received from the satellite. A 9-position selector switch permits selection of any combination of two pulses (of a possible nine combinations) to occur during the frame period.

2. Other Test Equipment

Tests and calibration equipment sufficient for maintaining and troubleshooting all units of the CDA ground stations is provided in each installation. In addition to the portable, general-purpose test instruments normally required for this type of installation, several rack-mounted units are provided. These units and their location and purpose are listed below:

- Frequency Counter. A Beckman-Berkeley, Model 7170, frequency counter, equipped with a Model 7570 video amplifier unit and Models 7571, 7572 and 7573 plug-in converter units, is located in rack 12. These units provide a frequency-measuring capability from 10 cycles per second to 1000 megacycles per second with an input sensitivity down to 1 millivolt.

- FM Signal Generator. An AN/URM-70 FM signal generator, used to test the TV receivers and the telemetry converters, is located in rack 12. This unit provides a 0.1- to 100,000-microvolt, frequency-modulated RF signal over a frequency range of 50 to 400 megacycles.

- Audio Oscillator. The audio oscillator located in rack 12 provides a sinewave signal variable from 18 cycles per second to 1.1 megacycles per second. The unit is used to calibrate demodulators, filters, tape recorders, etc.

- RF Power Measuring Equipment. Two units, a directional-coupler and a coaxial dummy load, are mounted in the command transmitter rack. The directional coupler, which is permanently connected in the output transmission line, is used to measure incident reflected power from which VSWR and radiated power are determined. The dummy load replaces the antenna for non-radiating tests.

In addition to the test instruments mentioned, several of the operating units of the ground station have self-testing capabilities.

SECTION IV. GO, NO-GO VAN

The Go, No-Go van, which is used for RF-interference testing and spacecraft checkout testing at the launch site during preparatory operations and during countdown, contains the necessary programming and transmitting equipment for commanding the spacecraft and the necessary receiving, demodulating, receiving and display equipment for checking spacecraft responses to the prescribed commands. The van contains essentially the same equipment used in the CDA stations with the exception that redundant units are not included nor are the units used for automatic operations or for antenna tracking operations.

PART IV. FIELD OPERATIONS

SECTION I. GROUND-SYSTEM OPERATIONS

The field operations performed by the CDA ground stations, i.e., those of tracking and commanding the satellite and, in turn, receiving, storing, and processing data transmitted from the satellite, were under direct NASA supervision for the ESSA 1 program. The operations, themselves, however, were similar to those performed on previous TIROS programs, and, in fact, almost identical to those performed on the TIROS IX program.

As on the previous TIROS programs, the Princeton ground station at the AED Space Center, Princeton, New Jersey, was utilized as a back-up facility. In addition, AED personnel were assigned to TTCC for the launch of ESSA 1 to advise and provide technical support for TTCC operating personnel in all operational-phase activities as required.

The prelaunch readiness countdown for ESSA 1 was begun on January 23, 1966. Daily circuit checks between TTCC and the CDA stations were performed with satisfactory results. A simulation exercise was begun on January 24 and successfully completed on January 26. The purpose of this exercise was to familiarize the CDA station personnel with the command sequences for the first 32 orbits of ESSA 1 as defined in the flight plan.

At the scheduled launch time the CDA stations and the Princeton station were in operational readiness. During the early orbits, the Princeton station was employed to acquire telemetry and attitude data from ESSA 1 when the satellite was commanded by the Wallops Island station. This data was monitored for any departure from nominal satellite conditions, as a back-up to a similar effort conducted at TTCC.

SECTION II. OPERATIONS AT CAPE KENNEDY LAUNCH SITE

Spacecraft OT-3 and the RF model were shipped from AED to the Cape Kennedy launch site at the Eastern Test Range (ETR) on January 17, 1966, and arrived there on the following day. After its arrival, spacecraft OT-3 was inspected to verify that it had not been damaged during shipment. A complete SEPET was started on January 19 and finished on January 20. Following completion of the SEPET, a fit check was performed with the flight-model attach fitting and showed that the mating surfaces closed properly.

On January 21, a Task IV was performed on spacecraft OT-3 and antennas were placed on the RF Model. Thereafter, the RF Model was transported to Hangar AM where it was mated with an inert third-stage rocket. The rocket/spacecraft combination was then taken to the launch pad.

On the following day, January 22, a second Task IV and a special camera resolution test were performed on spacecraft OT-3, and the solar cells on the hat assembly were cleaned. Also on January 22, a series of RF tests were started using the RF Model. These tests were completed on January 24, with no interference detected between the third-stage telemetry package and the spacecraft.

Spacecraft OT-3 was delivered to the spin balance facility on January 25 and mated with the third-stage of the launch vehicle. Although interference was noted between the bolt heads on the separation clamp and the antenna ring on the spacecraft, this was eliminated by milling down the bolt heads and by tying the nylon clamp lanyards more tightly.

On January 26, the spacecraft/third-stage combination was moved to launch pad 17A. Following installation of the combination on the second stage of the launch vehicle, the spacecraft was subjected to a full Task IV test. Immediately after the Task IV test, two command sensitivity checks were made with the tower in place; in one case, with the fairing on and, in the second case, with the fairing off. With the fairing off, there was no noticeable degradation of command sensitivity due to operation of the third-stage telemetry package; with the fairing on, the command sensitivity was reduced about 3 to 6 db by turning on the third-stage telemetry package. However, even in the worst-case condition, there was a command margin of approximately 30 db.

Complete Task IV tests were performed on January 27, January 29, and the morning of January 31. Late on January 31, the launch-day tests were started, with the launch scheduled for February 1.

Because of high winds in the upper atmosphere, on February 1, the launch effort was placed in a "hold" status at T-minus-82 minutes. The launch was rescheduled for February 3; the countdown on that date proceeded smoothly until T-minus-14 minutes, at which time a problem with a second-stage pressurization caused a "hold". A fault was found in the helium pressurization line, and corrected; after a 15-minute delay, the countdown proceeded normally, and lift-off took place at 074124Z. At 075419Z, ESSA 1 was successfully injected into orbit.

APPENDIX A.

ADMITTANCE MATRIX ANALYSIS OF THE 4-PORT HYBRID AT THE COMMAND RECEIVER INPUT

1. General

The following admittance matrix analysis of the 4-port hybrid used on the ESSA 1 satellite shows that the design is a perfectly matched 4-port hybrid when two ports are terminated in 50 ohms, and the other two are terminated in 25 ohms. The analysis also demonstrates that a failure resulting in a impedance change at the input to either command receiver does not affect the apparent source impedance seen by the opposite receiver. Figure A-1 is a schematic diagram of the 4-port hybrid.

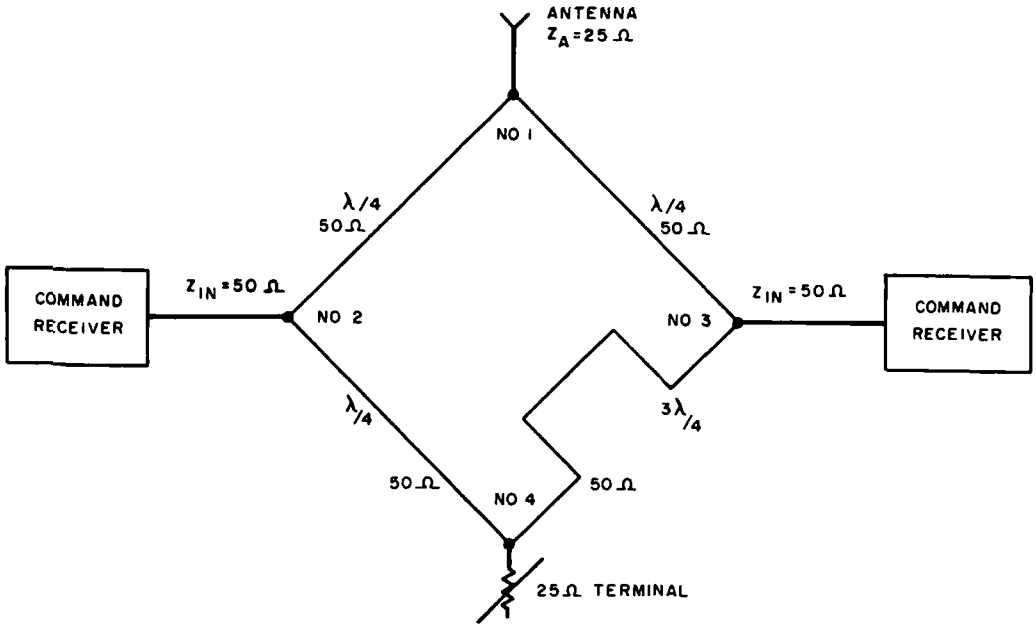


Figure A-1. 4-Port Hybrid, Schematic Diagram

2. Equivalent Circuit

Figure A-2 shows the equivalent circuit of the 4-port hybrid used in the analysis The equivalent circuit was obtained by replacing each arm of the 4-port hybrid by its π-

equivalent circuit. The conditions of the equivalent circuit are as follows:

$$Y_c = \frac{Y_o}{\sqrt{2k}}$$

where:

Y_c = characteristic admittance of transmission lines of the ESSA 1

$$Y_o = \frac{1}{25\,\Omega}$$

$$\frac{Y_o}{k} = \frac{1}{50\,\Omega}$$

$$k = 2$$

$$Y_c = \frac{Y_o}{2k} = \frac{Y_o}{2} = \frac{1}{50\,\Omega}$$

θ = electrical length of transmission lines.

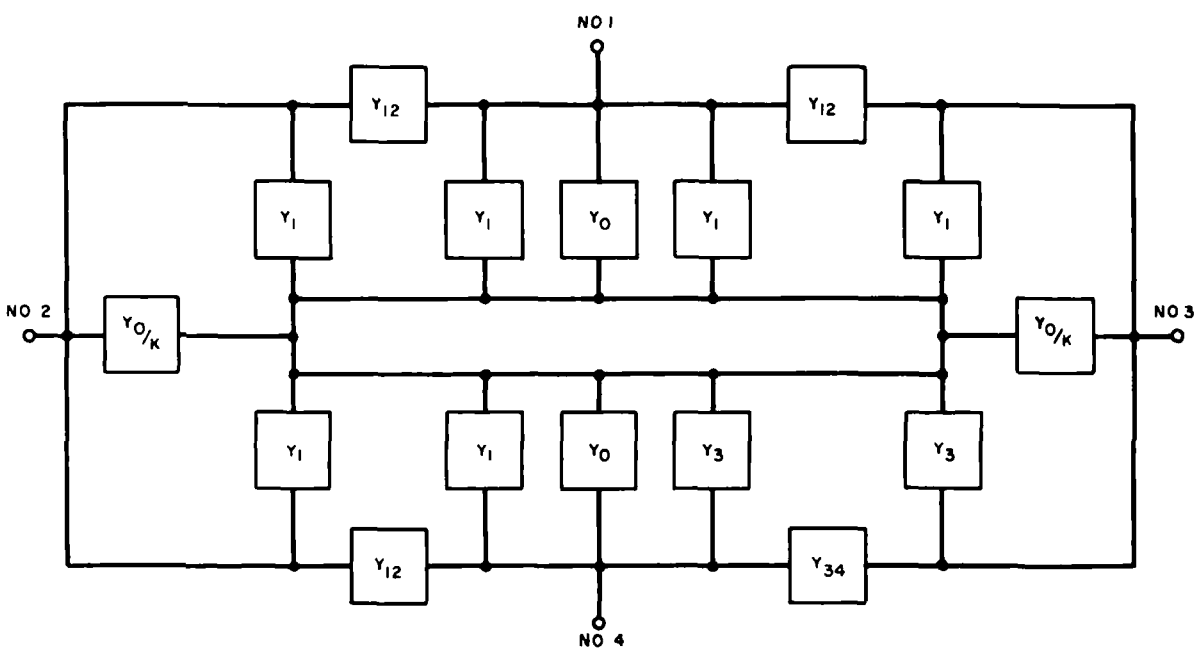


Figure A-2. Equivalent Circuit of 4-Port Hybrid

3. Input Admittance at Port No. 1 for Single Frequency at the Design Center of the 4-Port Hybrid

The following conditions exist for the special case of a fixed frequency operation at the design center of the 4-port hybrid, applied at port No. 1:

$$I = E_1 Y_{11} - E_2 Y_{12} - E_3 Y_{12} + 0$$

$$0 = -E_1 Y_{12} + E_2 Y_{22} + 0 - E_4 Y_{12}$$

$$0 = -E_1 Y_{12} + 0 + E_3 Y_{33} - E_4 Y_{34}$$

$$0 = 0 - E_2 Y_{12} - E_3 Y_{34} + E_4 Y_{44}$$

$$Y_{11} = Y_o + 2Y_1 + 2Y_{12}$$

$$Y_{22} = Y_o/k + 2Y_1 + 2Y_{12}$$

$$Y_{33} = Y_o/k + Y_1 + Y_3 + Y_{12} + Y_{34}$$

$$Y_{44} = Y_o + Y_1 + Y_3 + Y_{12} + Y_{34}$$

where:

$$Y_1 = jY_c \tan\frac{\theta}{2} \quad j\sqrt{\frac{Y_o}{2k}} \tan\frac{\theta}{2}$$

$$Y_{12} = -j\frac{Y_c}{\sin\theta} = -j\frac{Y_o}{\sqrt{2k \sin\theta}}$$

$$Y_3 = jY_c \tan\frac{3\theta}{2} = j\sqrt{\frac{Y_o}{2k}} \tan\frac{3\theta}{2}$$

$$Y_{11} = -\lambda \frac{Y_o}{\sin 3\theta} = -\lambda Y_o/\sqrt{2k \sin 3\theta}$$

$$\theta = \lambda/4 = 90°$$

$$Y_1 = jY_o/\sqrt{2k} \qquad Y_{11} = Y_o$$

$$Y_{12} = -jY_o/\sqrt{2k} \qquad Y_{22} = Y_o/k$$

$$Y_3 = -jY_o/\sqrt{2k} = -Y_1 \qquad Y_{33} = Y_o/k$$

$$Y_{34} = jY_o/\sqrt{2k} = -Y_{12} \qquad Y_{44} = Y_o$$

and:

$$I = E_1 Y_{11} - E_2 Y_{12} - E_3 Y_{12} + O$$

$$O = -E_1 Y_{12} + E_2 Y_{22} + O - E_4 Y_{12}$$

$$O = -E_1 Y_{12} + O + E_3 Y_{33} + E_4 Y_{12}$$

$$O = O - E_2 Y_{12} + E_3 Y_{12} + E_4 Y_{44}$$

Solving this determinant for E_4, we find:

(1) $E_4 = O$

(2) $E_2 = E_3$

} independent of generator impedance at port No. 1 or of terminating impedance at port No. 4.

The following simultaneous equations result:

$$I = E_1 Y_{11}' - 2E_2 Y_{12} \qquad (Y_{11}' = Y_{11} - Y_o)$$

$$O = -E_1 Y_{12} + E_2 Y_{22}$$

$$Y_{in} = \frac{I}{E_1} = Y_{11}' - \frac{2Y_{12}^2}{Y_{22}}$$

$$Y_{in} = O - \frac{2\left(\frac{-Y_o^{\,2}}{2k}\right)}{\frac{Y_o}{k}} = Y_o$$

$Y_{in} = Y_o$, independent of termination on port No. 4.

Therefore, a perfectly matched port independent of admittance on port No. 4 is obtained.

4. Input Match Looking Back into Port No. 2 (or Port No. 3)

To calculate the impedance looking back into the hybrid at port No. 2, assume port No. 1 to be terminated in a generator with output admittance Y_o.

$$O = E_1 Y_{11} - E_2 Y_{12} - E_3 Y_{12} + O$$

$$I_2 = -E_1 Y_{12} + E_2 Y_{22}' + O - E_4 Y_{12}$$

$$O = -E_1 Y_{12} + O + E_3 Y_{33} + E_4 Y_{12}$$

$$O = O - E_2 Y_{12} + E_3 Y_{12} + E_4 Y_{44}$$

Solving this determinant for E_3,

(1) $E_3 = O$

(2) $E_1 = E_4$

} independent of generator impedance at port No. 2 or of terminating impedance at port No. 3.

$$O = E_1 Y_{11} - E_2 Y_{12}$$

$$I_2 = -2E_1 Y_{12} + E_2 Y_{22}' \qquad \left(Y_{22}' = Y_{22} - \frac{Y_o}{k}\right)$$

$$Y_{in_2} = \frac{I_2}{E_2} = Y_{22}' - \frac{2Y_{12}^2}{Y_{11}}$$

$$Y_{in_2} = O - \frac{2\left(\frac{-Y_o^2}{2k}\right)}{Y_o} = \frac{Y_o}{k}$$

$Y_{in_2} = \frac{Y_o}{k}$, independent of termination on port No. 3.

Therefore, a perfectly matched port independent of admittance on port No 3 is obtained.

APPENDIX B.

RF SYSTEM ANALYSIS FOR SPACECRAFT OT-3

1. Beacon Transmitter Modulation Levels

a. General

In establishing satisfactory modulation levels for the spacecraft OT-3 beacon transmitters, it was necessary to consider several related effects, as follows:

(1) Effect on the signal-to-noise ratio at the receiver output at the CDA station.

(2) Effect on the signal -to-noise ratio at the FM demodulator output at the CDA station.

(3) Effect on the spurious RF signal levels on board the spacecraft.

b. Effect on Receiver Output Signal-to-Noise Ratio

The relationship for AM signal-to-noise ratio in a linear system is given by:

$$S/N_{AM} = C/N\ M^2\ B_I/2F_B, \qquad (1)$$

where,

C/N is the receiver carrier-to-noise ratio,

M is the modulation index,

B_I is the information bandwidth, and

F_B is the post-detection bandwidth.

As can be seen, in a linear system, the maximum signal-to-noise ratio occurs at the maximum modulation index. However, when the modulator is non-linear, as in the case being considered, the effect is as shown in Figure B-1. Whereas the fundamental signal continues to increase with increased modulation index, the rate of increase is

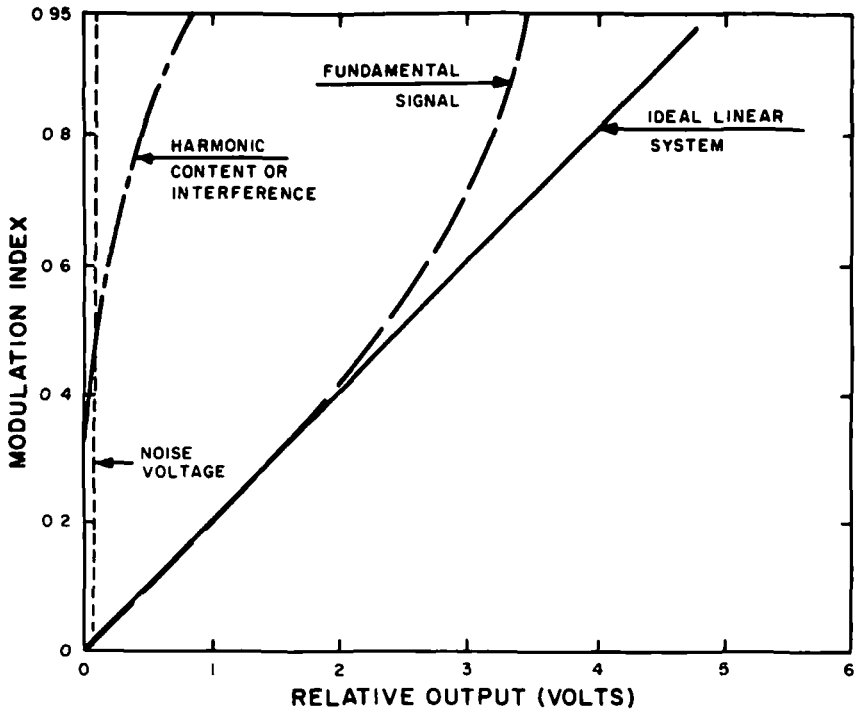


Figure B-1. CDA Ground Station Receiver Output vs Satellite Transmitter Modulation Index

drastically reduced and the harmonic content rises out of the noise. The first detrimental effect of this nonlinearity is felt on the ground in the receiver AGC network. The peak detection and integrating circuits of the AGC network act on the composite audio signal consisting of desired signal plus interfering harmonics, and thereby tend to establish a false AGC level for the desired signal. Since the AGC level fixes the energy of the receiver output, the contribution of the desired signal to the total energy is diminished. Therefore, in a non-linear system, the increased modulation index will not increase the AM signal-to-noise ratio as specified in equation (1).

c. Effect on FM Demodulator Output Signal-to-Noise Ratio

The FM demodulator output is distorted by interfering signals caused by cross-talk between the channels of a given beacon and introduced in the AM detection from both beacons. Therefore, both the spurious RF content and the harmonic audio content introduce signals within the channel band-width allocated to the FM demodulator. The degree of the modulation distortion can be readily calculated, as follows:

$$m = \sqrt{\frac{(uc)^2 + (uc_2)^2 + (uc_3)^2}{AD}}$$

where

m is the degree of modulation (percent modulation),

u is the signal frequency minus the interference frequency,

c is the amplitude of the interfering audio signal*,

D is the deviation, and

A is the signal amplitude.

An example of this modulation distortion calculation for a single interfering signal that is 20 db below the desired carrier and differs in frequency from the signal frequency by the low-pass output cutoff frequency, 50 cps, is as follows:

Channel 5

$$m = \sqrt{\frac{(50 \times 0.1)^2}{100}} = 5 \text{ percent}$$

Channel 7

$$m = \sqrt{\frac{(50 \times 0.1)^2}{172}} = 5 \text{ percent}$$

Thus, the amplitude of all harmonic content within the bandpass of the demodulator must be minimized to improve the output signal-to-noise ratio. To achieve this goal, the source of the detrimental harmonic generation, the non-linear modulator, should be operated at a modulation index which will reduce the harmonic content. For spacecraft OT-3, the modulation levels were established on the basis of measured spurious output at different modulation levels. The spacecraft OT-3 modulation percentages were established as follows:

- High Beacon: 80 to 85 percent
- Low Beacon: 70 to 75 percent

d. Effect on the Spurious RF Signal Levels on the Spacecraft

In addition to reducing energy from the carrier and causing additional noise input to the ground receiver, the undesirable spurious RF can reduce spacecraft command sensitivity. Spurious frequencies falling within the command system bandpass can easily capture the system. In addition, energy emitted at frequencies adjacent to the energy bandpass can overcome the IF selectivity if sufficient power is available. It is possible

*c is reduced when u is higher than the low-pass cutoff frequency of the FM demodulator output.

to operate on the command receiver AGC unless steps are taken to increase the isolation between the beacon and command subsystems. The spurious energy level of the beacon transmitter is a function of the modulation index and is reduced as the percent of modulation is decreased. The transmitters were carefully tested to determine the modulation-level setting that initiated spurious outputs. In this test, the performance of the high-frequency beacon selected for spacecraft OT-3 was exceptionally good; the unit tolerated a 90-percent modulation without producing excessive spurious output. The low-frequency beacon transmitter selected was susceptible to spurious generation beginning at a level of 70 percent.

2. Margin and Path-Loss Calculations for the Modified Spacecraft OT-3 RF System

The received power at the Princeton ground station for the modified RF system was calculated to be as follows:

Transmitter (70 mw):	+ 17.8 dbm
Satellite Antenna Gain:	+ 2.7 db
Princeton Ground Station Antenna Gain:	+ 18.0 db
136-Mc Filter and Pad:	- 4.8 db
Path loss @ 950 NM @ 5° elevation:	-144.2 db
Line Loss:	- 1.0 db
Received Power:	-117.4 dbm

Noise power in a 16-kc IF amplifier with a 4.5-db noise figure is -125.2 dbm. Therefore, the carrier-to-noise ratio is +7.8 db and a 0.8-db margin exists above a 7-db AM receiver threshold. If the 8-kc IF amplifier is used on the receiver with tuning, an additional 3-db margin is available.

$$S/N_{AM} = C/N\ M^2\ B_i/2F_B$$

For the worst case (Channel 7, low-frequency beacon):

M = 0.35

B = 344 cps

f_m = 16 kc

Therefore, S/N_{AM} = 7.8 db + 20 log 0.35 + 10 log 23.3,

= 7.8 db - 9.12 + 13.67,

= 12.4 db.

The improvement threshold for the FM demodulator is 12 db, and therefore, a 0.4-db margin exists.

For the Princeton ground station, a zero margin exists at the 5-degree elevation angle, and the margin increases at a rate of approximately 1 db per each additional 5 degrees of antenna elevation. For channel 5 on the high frequency beacon a 3-db margin normally exists at the input to the FM demodulator.

For the Wallops Island and Fairbanks CDA ground stations, the available antenna gain and noise figure are as follows:

	Antenna Gain	Noise Figure
Wallops Island	22 db	-
Fairbanks	26 db	4.0

Thus, the operational stations have an additional margin of approximately 4 db at the 5-degree elevation angle of the ground antenna, limited by local station obstructions.

3. Isolation of Command Receiver from Beacon Transmitter Spurious Signals

The worst case for spurious beacon input to the command receivers is as follows:

Worst-Case Spurious Power from Beacons:	- 10 dbm
BP Filter Loss at 148 Mc:	- 35 db
Pad Loss:	- 3 db
Antenna Losses:	- 28 db
Receiver Power Split:	- 3 db
	- 79 dbm

The worst case spurious beacon power occurs only at high modulation levels. The losses will vary widely depending on the spurious frequency due to the tuning characteristics of both filters; however, the command receiver IF selectivity offers

rejection to approximately -47 dbm over a broad range with some discrete responses at -65 dbm. With the reduction in modulation levels and increased isolation, there exists approximately 13 db or more margin for cases of on-board receiver desensitization.